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PERFORMANCE SUMMARY
FOR THE

# JASON

SOUNDING ROCKET VEHICLE

REPORT NO. AST/E1R-13325

18 April 1961

This report was prepared by Vought Astronautics, a Division of Chance Vought Corporation, Dallas, Texas, under Contract No. NAS1-1013 administered by NASA, Langley Research Center.

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Program Manager –
Booster Systems

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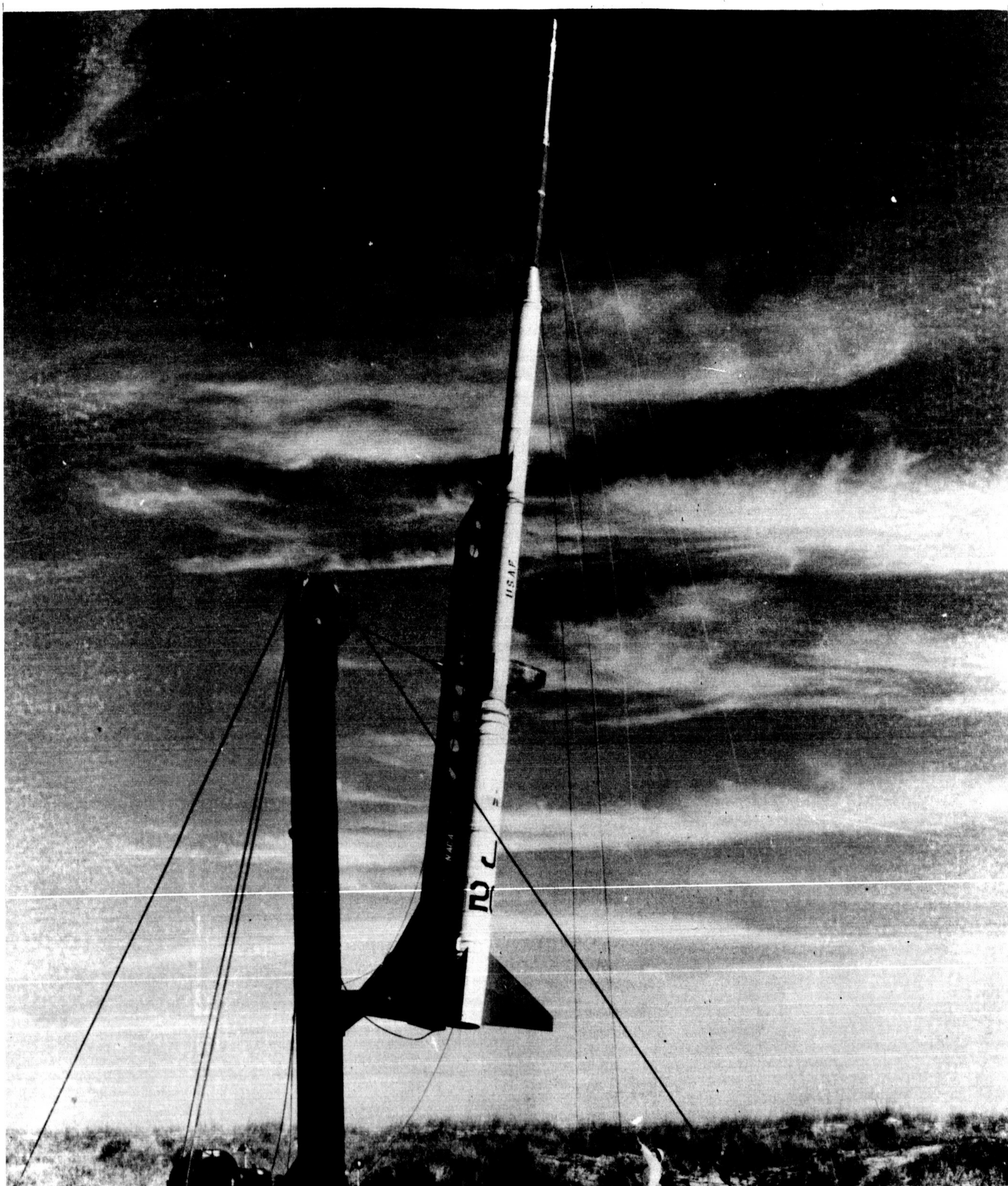
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# FOREWORD

The primary purpose of this report is to aid in the preliminary selection of a vehicle for a specific payload mission. Performance data in this report show a broad flight regime and have not been modified by restraint items such as aerodynamic heating, range safety and other detail factors. In fact, this cannot be done until a mission has been established. Thus, caution must be used in extracting detail data. It is believed that the information presented will allow the user to consider all the major aspects of the booster system, and will serve as a guide in payload system integration.

# INTRODUCTION

This Sounding Rocket Handbook report is one of a series prepared by Vought Astronautics, a Division of Chance Vought Corporation, for the National Aeronautics and Space Administration under Contract No. NAS1-1013. This contract was administered by the Langley Research Center under the technical direction of Hal T. Baber, Jr., of the Vehicle Performance Branch, Applied Materials and Physics Division, Langley Research Center. This report presents data for one of the eighteen vehicle systems listed below:

| Vehicle | Handbook No. | Vehicle | Handbook No. |
|---|---|---|---|
| Aerobee 100 | AST/E1R-13318 | Journeyman | AST/E1R-13327 |
| Aerobee 150A | AST/E1R-13319 | Journeyman B | AST/E1R-13328 |
| Aerobee 300A | AST/E1R-13320 | Jaguar | AST/E1R-13329 |
| Arcas | AST/E1R-13321 | Little Joe | AST/E1R-13330 |
| Arcon | AST/E1R-13322 | Nike-Asp | AST/E1R-13331 |
| Exos | AST/E1R-13323 | Nike-Cajun | AST/E1R-13332 |
| Iris | AST/E1R-13324 | Shotput | AST/E1R-13333 |
| Jason | AST/E1R-13325 | Skylark | AST/E1R-13334 |
| Javelin | AST/E1R-13326 | Strongarm | AST/E1R-13335 |

In addition to the handbooks on each vehicle, the following handbooks have been prepared:

| Handbook | Handbook Number |
|---|---|
| Summary Report | AST/E1R-13337 |
| Rocket Motor Ballistic Data Report | AST/E1R-13336 (Confidential) |
| Cost and Reliability Summary | AST/E1R-13338 |

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# VEHICLE DESCRIPTION

## Summary

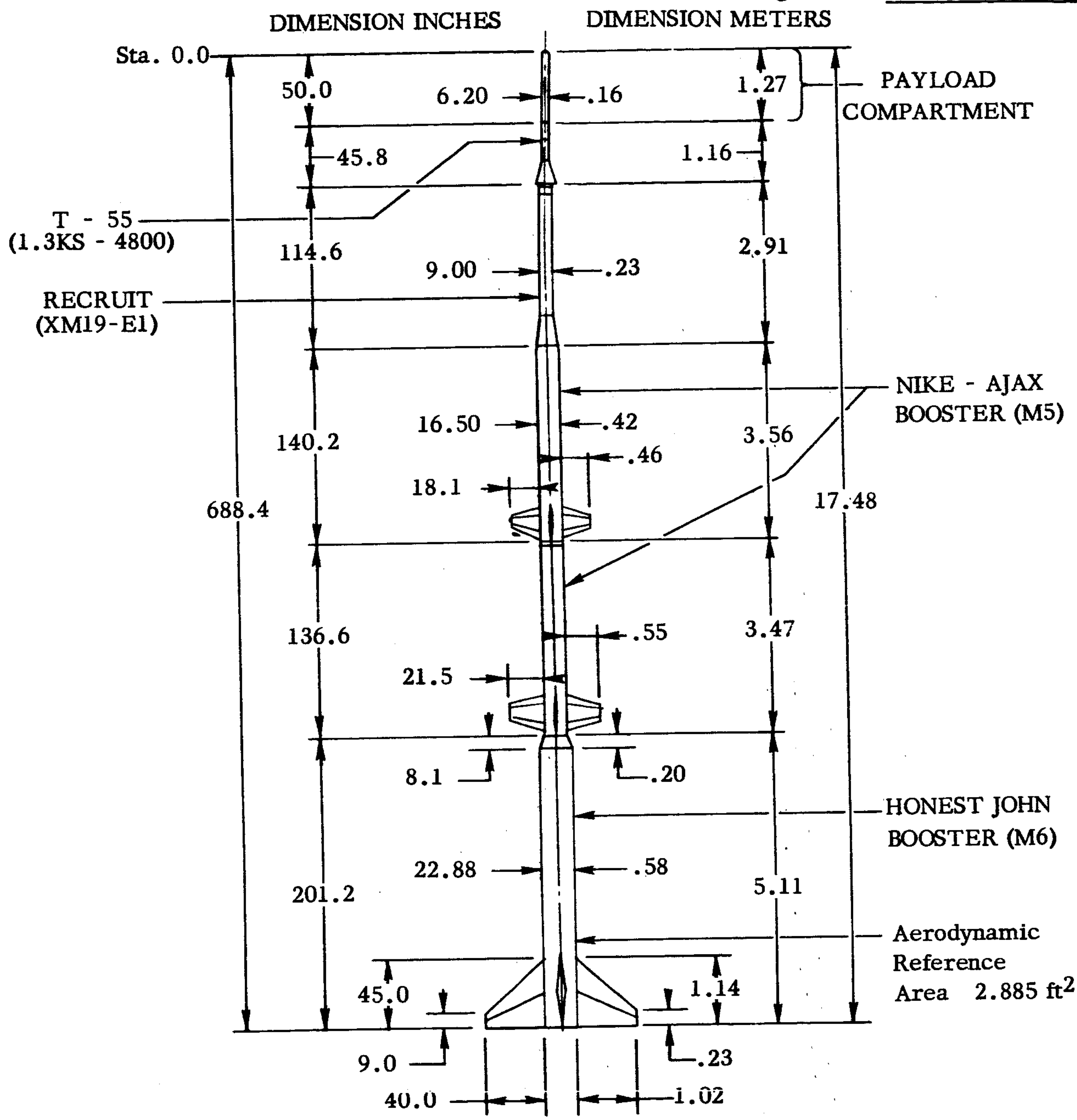
| | |
|---|---|
| NAME OF VEHICLE | JASON |
| DESIGNATION | ARGO E-5 |
| MANUFACTURER | AEROLAB DEVELOPMENT COMPANY |
| NUMBER OF STAGES | 5 |
| LAUNCH WEIGHT (No Payload) | 7242.2 POUNDS |
| OVER-ALL LENGTH | 688.4 INCHES |
| MAXIMUM DIAMETER | 22.88 INCHES |
| PRIME USERS | NASA<br>AIR FORCE |
| NET PAYLOAD | |
| NOMINAL | 40.0 POUNDS |
| MINIMUM | 20.0 POUNDS |
| MAXIMUM | 60.0 POUNDS |
| VOLUME | .8 CUBIC FEET |
| PERFORMANCE AT NOMINAL NET PAYLOAD | |
| APOGEE ALTITUDE (VERTICAL LAUNCH) | 640 NAUTICAL MILES |
| ACCELERATION, MAXIMUM | 125 "g" |

The vehicle assembly and staging weight is shown in Figure 1. Figure 2 shows the aerodynamic outline of the payload compartment used in the analysis and the associated usable volume.

## Background

The Jason is a five-stage, aerodynamically stabilized sounding rocket utilizing solid propellant rocket motors. Fins provide static stability on the first, second, and third stages. The fourth and fifth stages are stabilized by means of flared sections at the aft end of each stage. The rocket motors used are Honest John, first step; Nike-Ajax Booster, second and third steps; Thiokol XM 19E1 Recruit, fourth step; and Thiokol T-55, fifth step. The first stage is ignited at launch and burns for approximately 4 seconds. At burnout the expended first step drag separates, and the vehicle coasts for about 5 seconds to second stage ignition. Second stage burning time is about 3.5 seconds with drag separation at burnout. A 22 second coast to third stage ignition follows. Third stage burning time is 3.5 seconds, and the expended third step remains attached through approximately 1.5 seconds of coast. Step three separation is effected by rupture of a NASA blowout diaphragm at fourth stage ignition. Stage four burning time is about 3 seconds, with immediate ignition of stage five for about 3 seconds of burning. Diaphragm separation is also used for step four. The Jason has been operationally fired 19 times with 89.5% reliability. The Jason has been fired from both a modified I-beam (Honest John) and tubular (Sergeant) launcher.

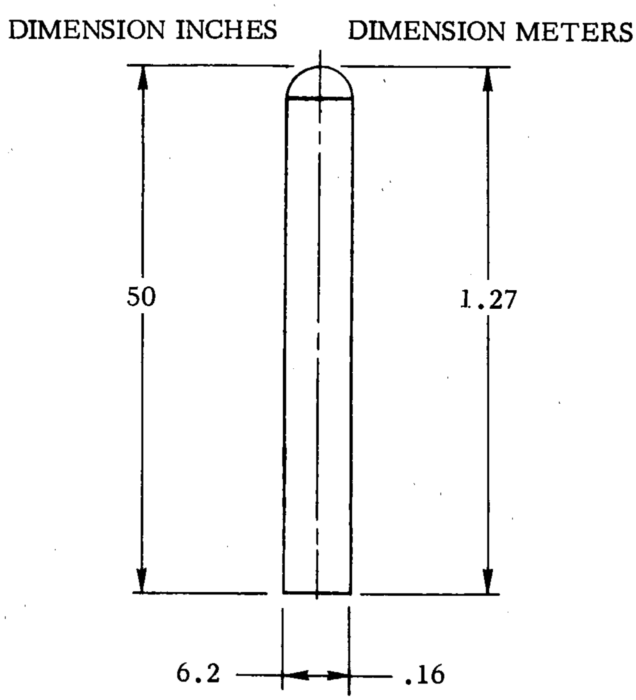
JASON




PERFORMANCE WEIGHTS LESS PAYLOAD

| POUNDS | | KILOGRAMS |
|---|---|---|
| 7242.2 | Launch | 3285.0 |
| 5062.2 | B. O. 1st STAGE | 2296.2 |
| 3072.2 | Fire 2nd STAGE | 1393.5 |
| 2308.2 | B. O. 2nd STAGE | 1047.0 |
| 1749.9 | Fire 3rd STAGE | 793.7 |
| 985.9 | B. O. 3rd STAGE | 447.2 |
| 444.9 | Fire 4th STAGE | 201.8 |
| 176.9 | B. O. 4th STAGE | 80.2 |
| 62.6 | Fire 5th STAGE | 28.4 |
| 29.4 | B. O. 5th STAGE | 13.4 |

FIGURE 1 VEHICLE ASSEMBLY AND STAGING WEIGHTS




NOMINAL AVAILABLE PAYLOAD VOLUME =
.8 CU. FT. (.023 CU. METERS)

FIGURE 2 PAYLOAD COMPARTMENT

# FLIGHT PERFORMANCE

The flight performance data presented in this study show a very broad flight regime for each vehicle. Modifications to the data have not been made to account for factors such as the launch site, launcher elevation limits, range safety, and vehicle-payload environment. Consideration of these factors usually results in limitations being placed on the flight regime. Some limitations may be removed by minor modifications, while in the case of range safety, the limitation may be revised with no modification as the vehicle builds a good operational history. If the flight performance data were based on a set of firm limits, it would be very difficult to extrapolate the data. However, it is rather easy to restrict, when necessary, the broad flight regime shown in Figures 3 through 19. Some degree of caution must be exercised in interpreting these figures. For example, the vehicle was considered to be a "clean" aerodynamic configuration, i.e., it was assumed to have no external antennae, even though certain experiments in the past may have been flown with antennae. Further, all performance is presented for net payload, as defined in the Nomenclature.

Flight performance calculations were conducted with an IBM 704 digital computer using two degree of freedom analysis on a spherical, non-rotating earth. The routine considered aerodynamic coefficients to be Mach number-dependent, while thrust was computed by correcting time-dependent vacuum thrust for ambient pressure. The 1959 ARDC model atmosphere was used. Coasting trajectories with earth rotation and oblateness were computed using a different IBM routine. Burnout was assumed to occur at 35° North latitude and 285° East longitude, at an azimuth of 135°. The two degree of freedom last stage burnout conditions were used as input data in an axis system which rotates with the earth.

## Trajectory

Actual gravity turn (sometimes called zero lift) trajectories were calculated for the Jason vehicle at launch angles of 70, 80, and 88 degrees, each at net payloads of 20, 40, and 60 pounds. The launch angles were chosen to show a very broad flight regime, while the payloads were estimated to be minimum, nominal, and maximum. Range to impact information for the expended steps was determined using an approximate drag coefficient for an arbitrarily oriented body. It is recognized that an expended step may have a preferred, lower drag, orientation. Impact range for those steps which operate in the denser atmosphere might, in reality, be greater than shown in this report. In any case, range safety is of such importance that detailed study is required.

## Dispersion

Jason dispersion data are included in Reference 2, but to achieve consistency with the other vehicles in the study, calculations were conducted on the IBM 704 digital computer, using the results of the performance calculations as a starting point. The following values were used as one sigma variations at burnout:

a. Pitch flight path angle, $\pm 2^\circ$
b. Yaw flight path angle, $\pm 2^\circ$
c. Velocity, $\pm 1$ per cent

Trajectories were computed from burnout to impact for each of these conditions for an 85° launch angle and a nominal payload. Dispersion was then calculated as the root mean square of the individual contributions. The dispersion radii for steps one, two, three, four, and five are thus approximately .3, 1., 8., 43. and 70. nautical miles.

The dispersion data presented here are too small if arbitrary winds at launch are considered. Since wind dispersion can be a very serious problem for an unguided vehicle of this type, detailed study would be required.

## Actual and Ideal Velocity

Incremental ideal and actual velocity as a function of payload are shown in Figure 20 and Figure 21, while Figure 22 shows both this data, at a nominal payload, and the velocity losses due to drag and gravity. All this information is presented for an 88° launch angle. Incremental actual velocity was obtained directly from the computer runs. Incremental ideal velocity was computed in the standard manner:

$$\Delta V_{ID} = (I_{sp})_{AVG}\, g_s \ln \mu$$

where average specific impulse was determined by integration of the thrust-time trace from the computer runs, and dividing the result by the consumed weight.

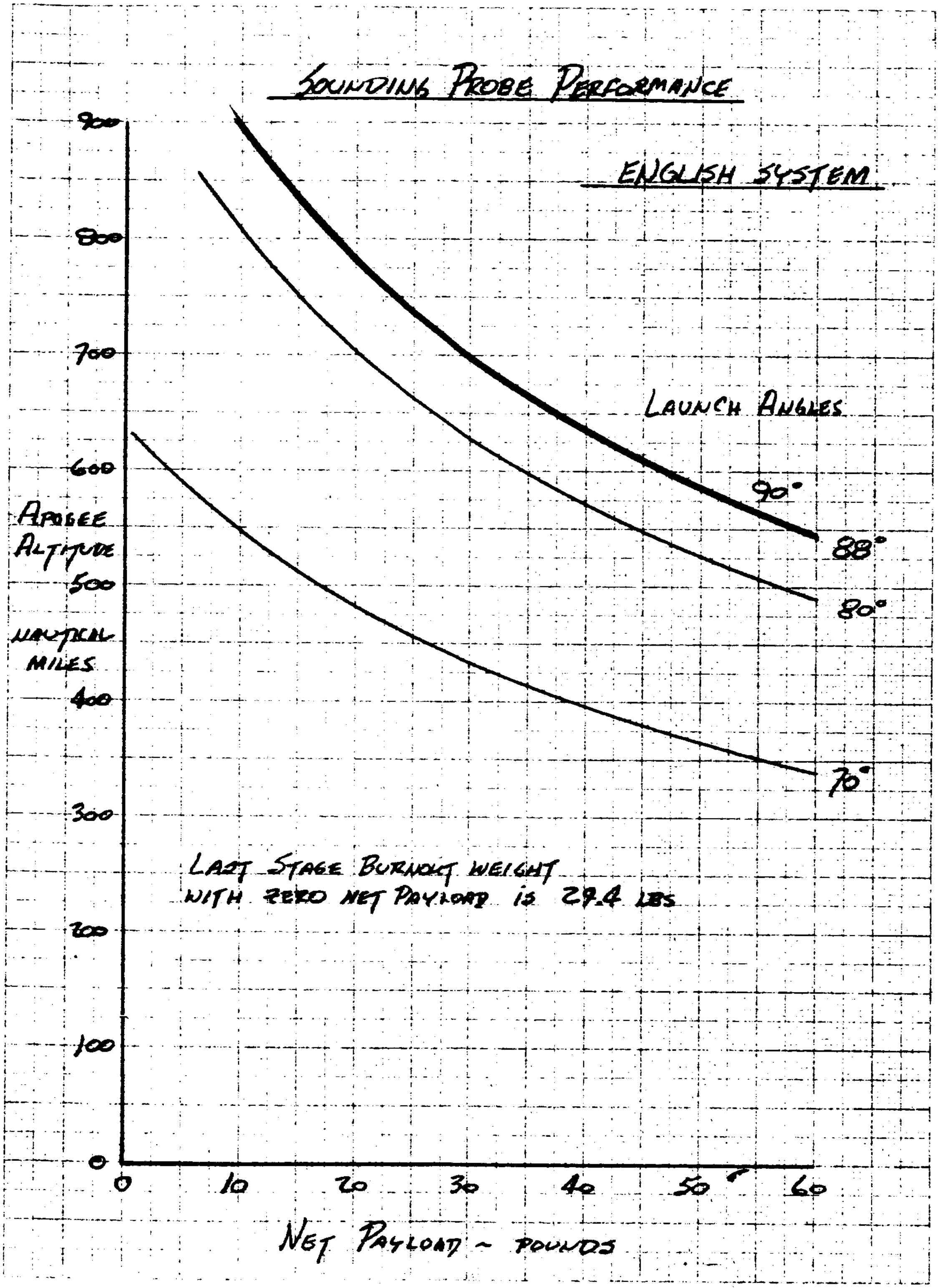



FIGURE 3 SOUNDING PROBE PERFORMANCE (ENGLISH SYSTEM)

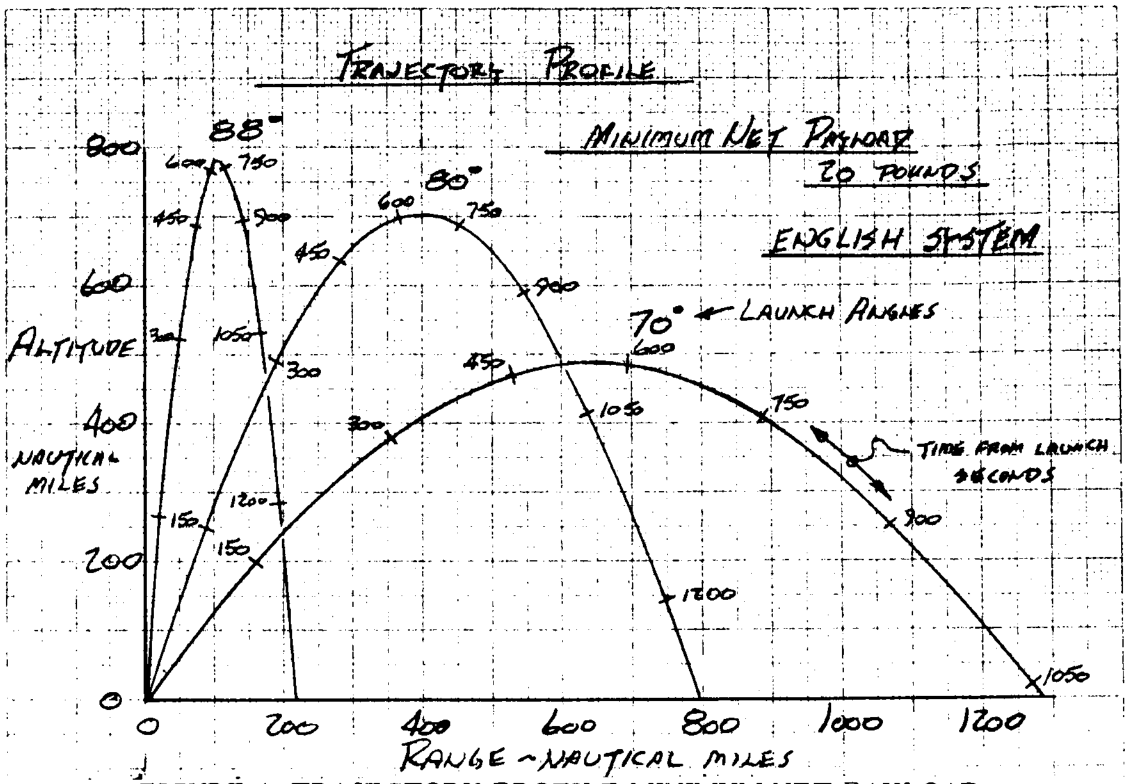



FIGURE 4 TRAJECTORY PROFILE MINIMUM NET PAYLOAD 20 POUNDS (ENGLISH SYSTEM)

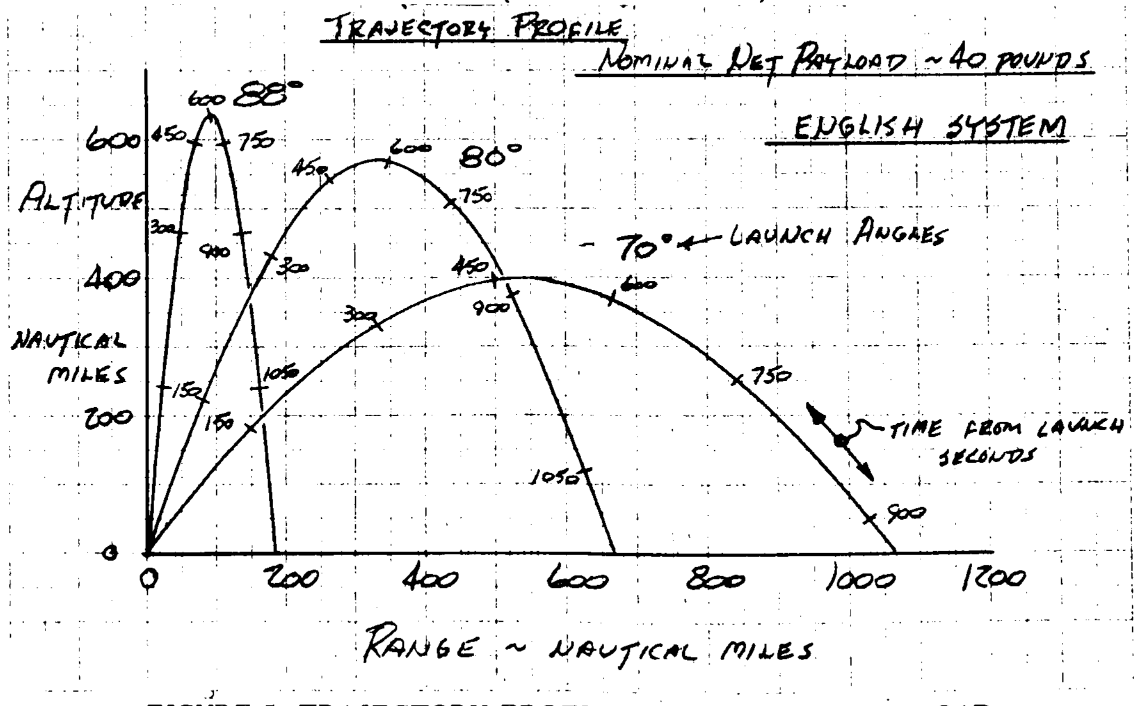



FIGURE 5 TRAJECTORY PROFILE NOMINAL NET PAYLOAD 40 POUNDS (ENGLISH SYSTEM)

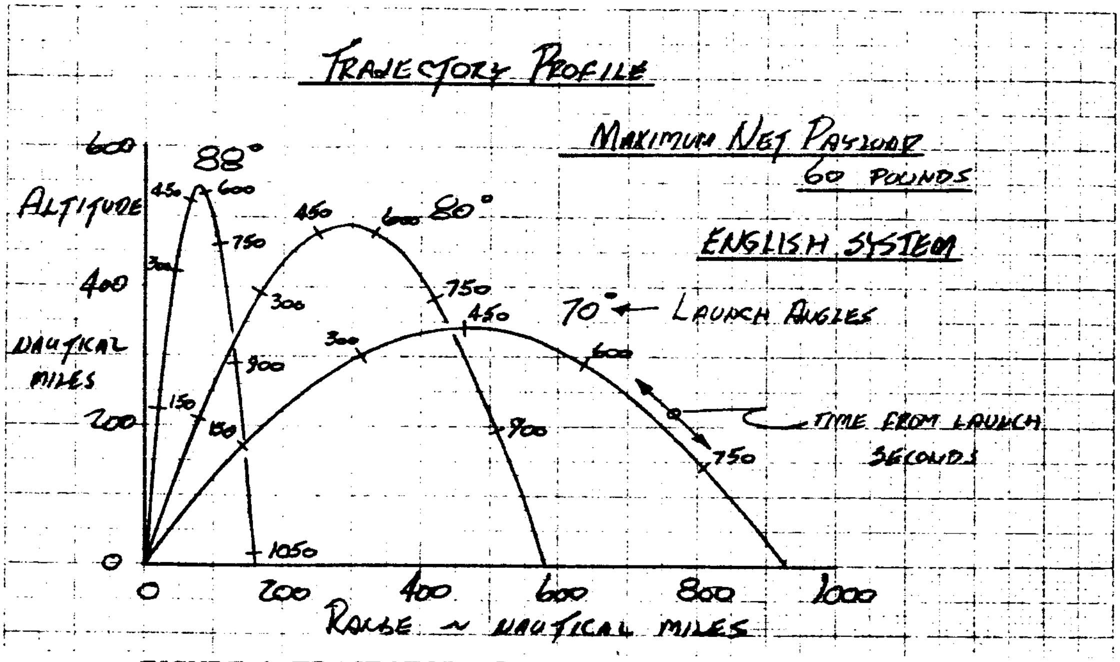



FIGURE 6 TRAJECTORY PROFILE MAXIMUM NET PAYLOAD 60 POUNDS (ENGLISH SYSTEM)

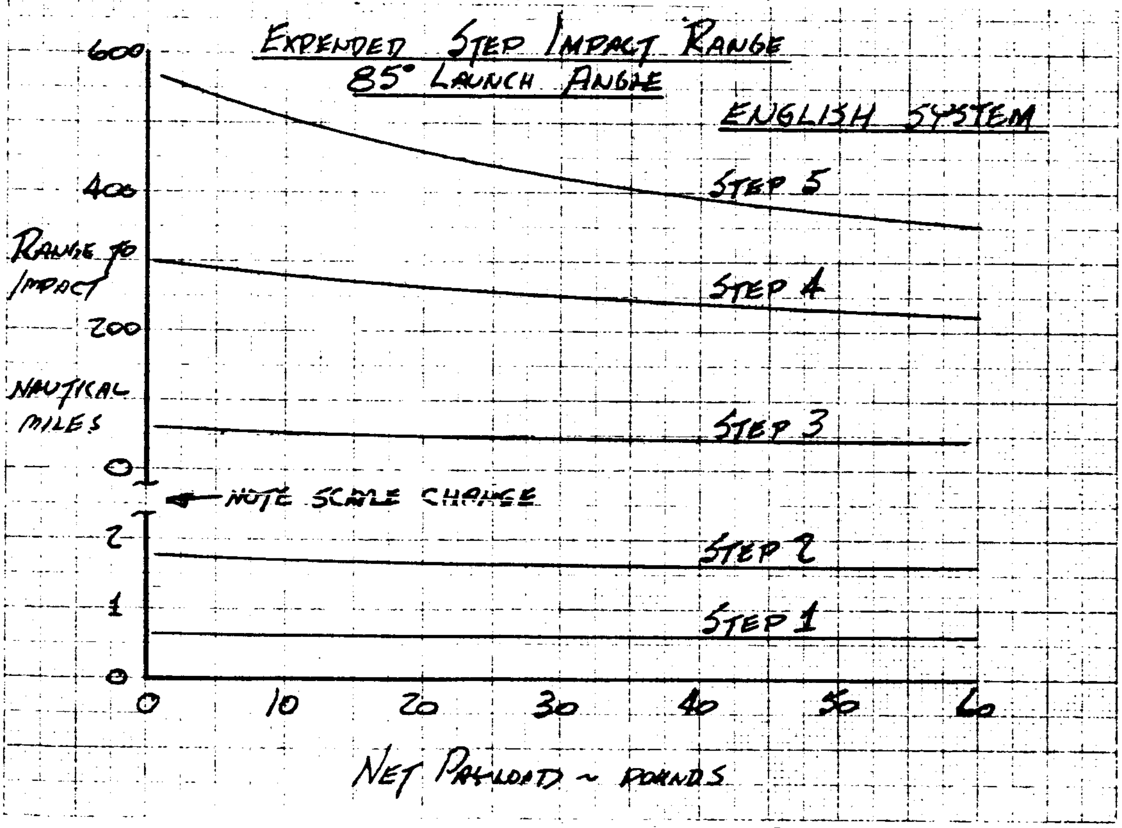



FIGURE 7 EXPENDED STEP IMPACT RANGE $85^{O}$ LAUNCH ANGLE (ENGLISH SYSTEM)

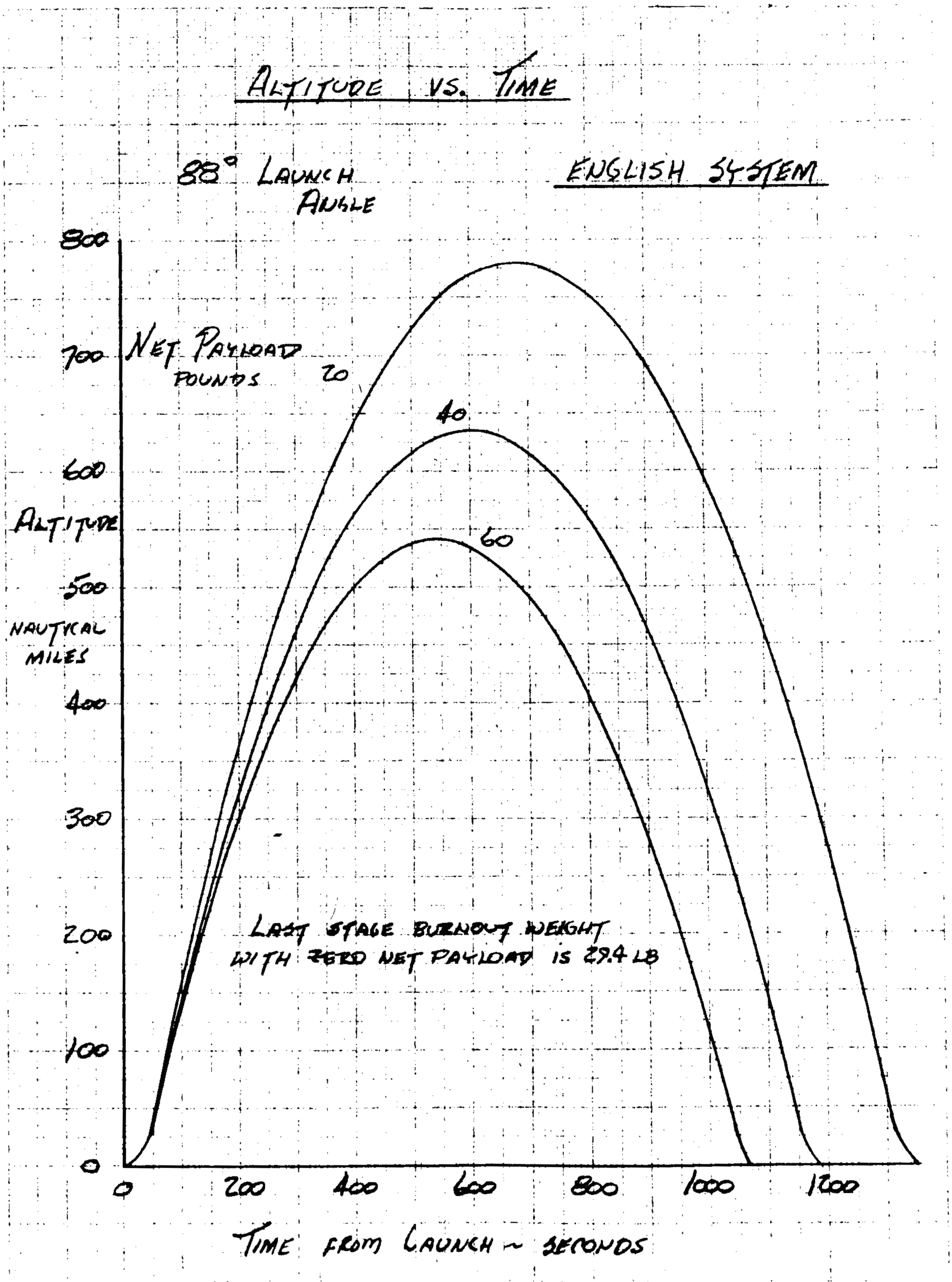



FIGURE 8 ALTITUDE Vs. TIME (ENGLISH SYSTEM)

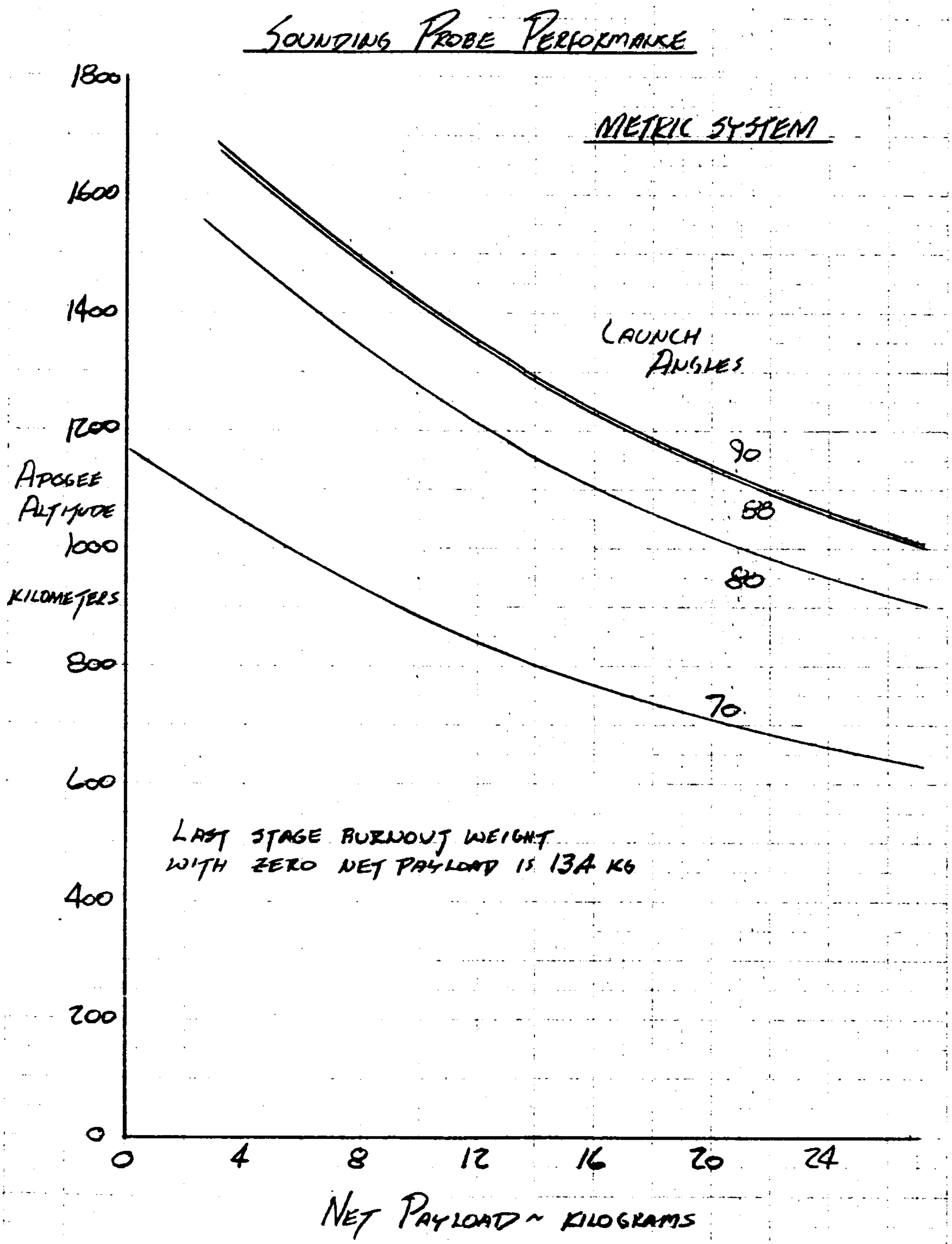



FIGURE 9 SOUNDING PROBE PERFORMANCE (METRIC SYSTEM)

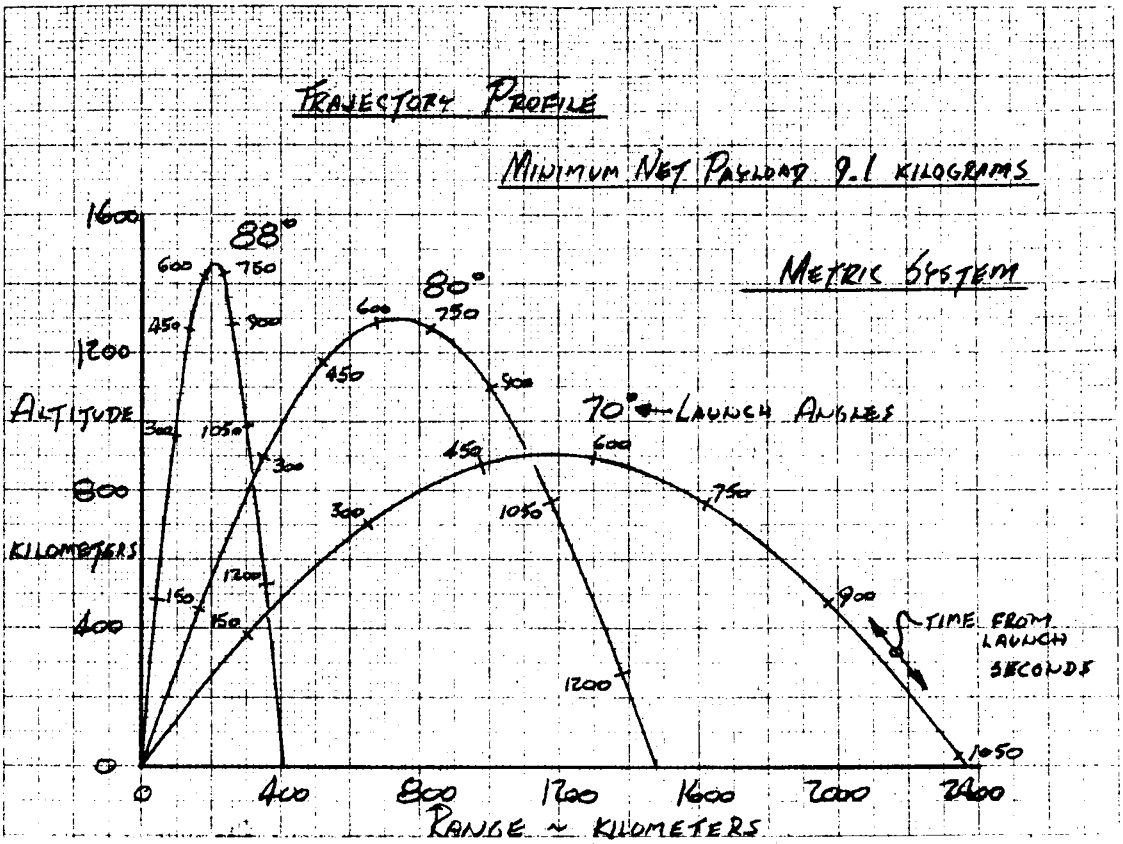



FIGURE 10 TRAJECTORY PROFILE MINIMUM NET PAYLOAD 9.1 KILOGRAMS (METRIC SYSTEM)

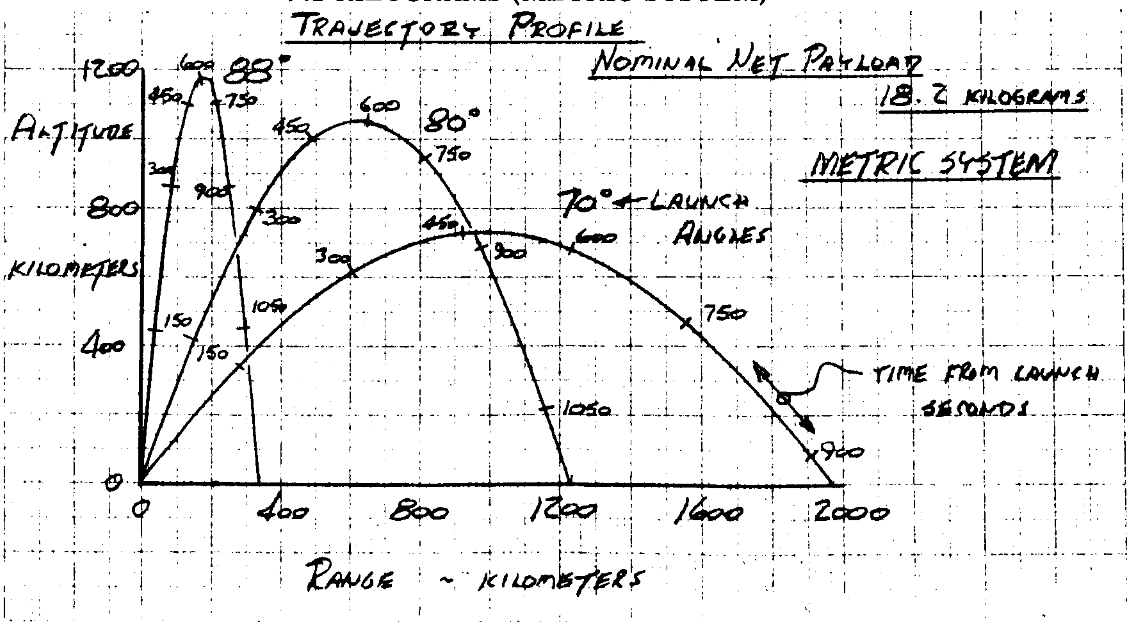



FIGURE 11 TRAJECTORY PROFILE NOMINAL NET PAYLOAD 18.2 KILOGRAMS (METRIC SYSTEM)

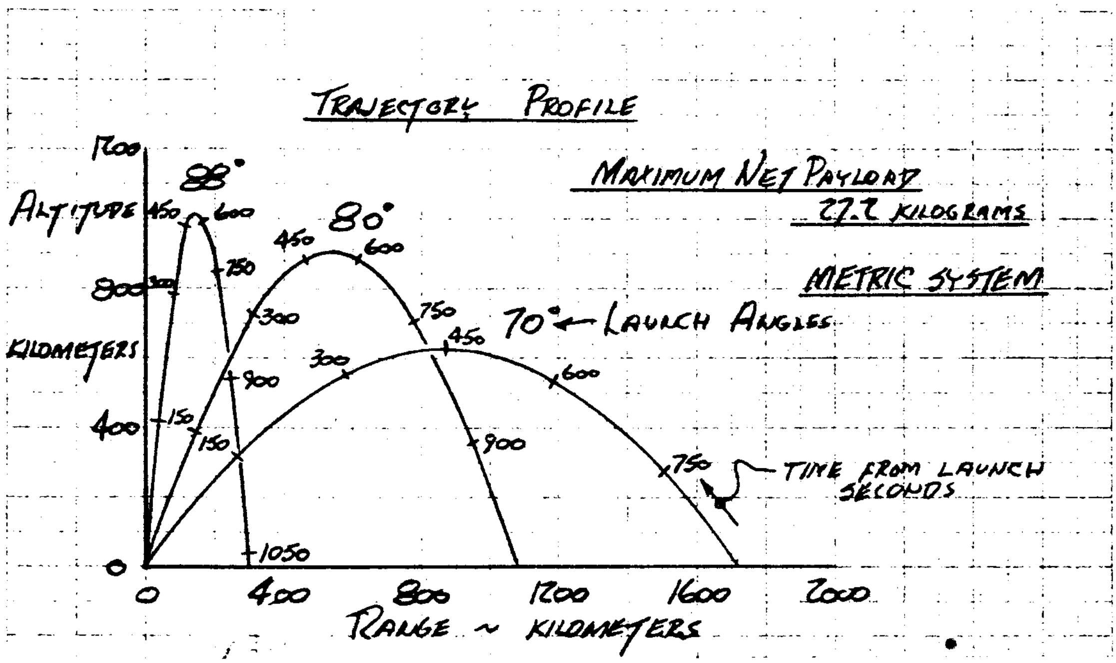



FIGURE 12 TRAJECTORY PROFILE MAXIMUM NET PAYLOAD
27.2 KILOGRAMS (METRIC SYSTEM)

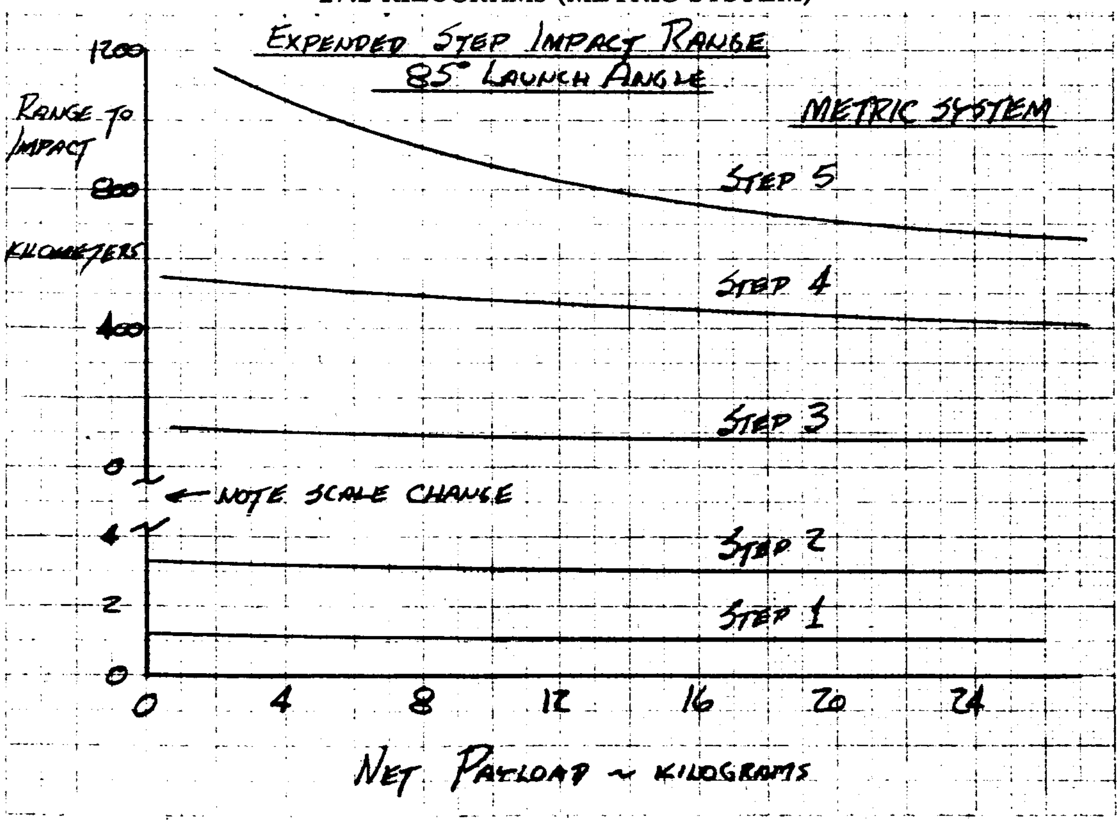



FIGURE 13 EXPENDED STEP IMPACT RANGE 85° LAUNCH ANGLE
(METRIC SYSTEM)

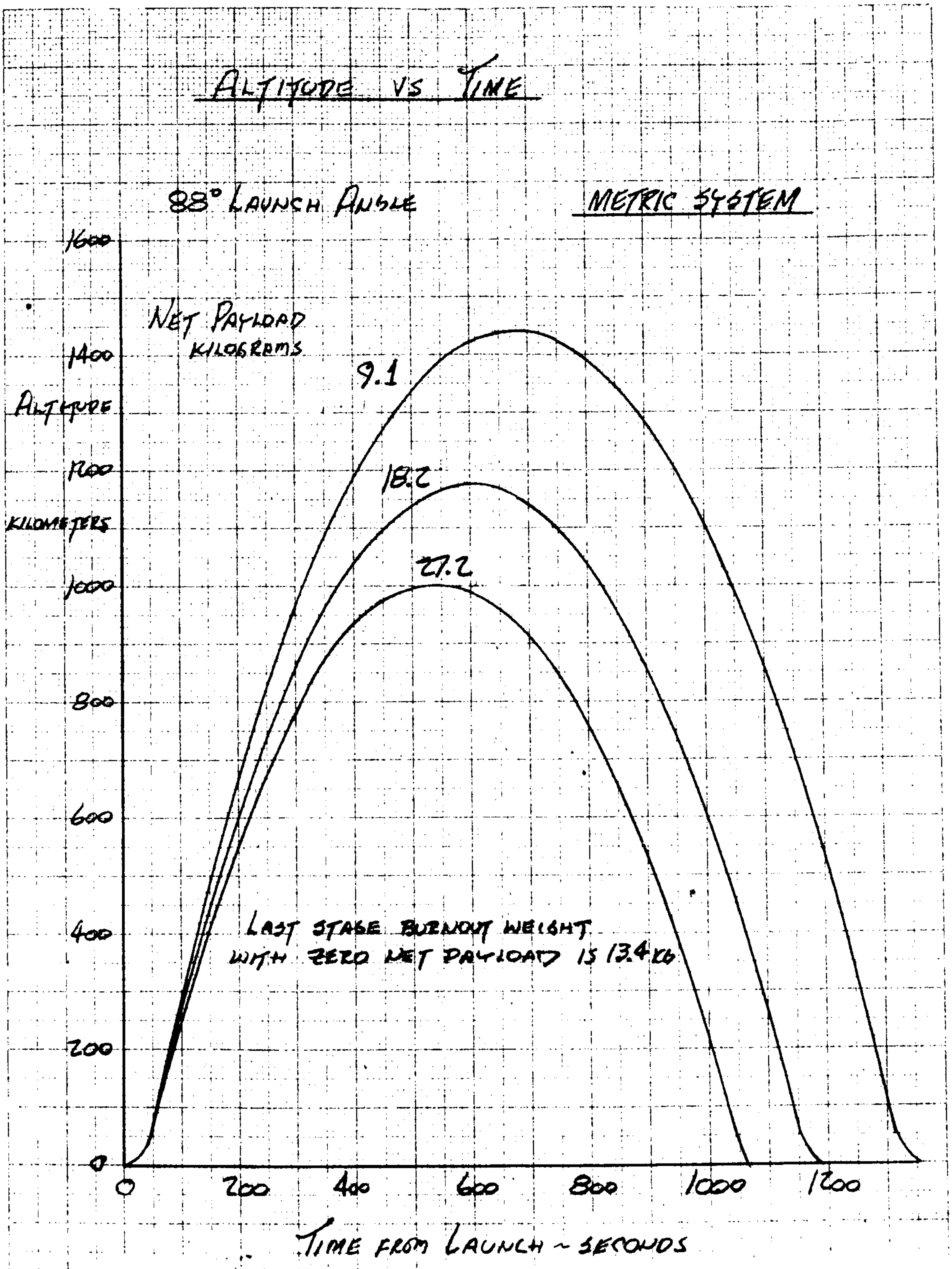



FIGURE 14 ALTITUDE Vs. TIME (METRIC SYSTEM)

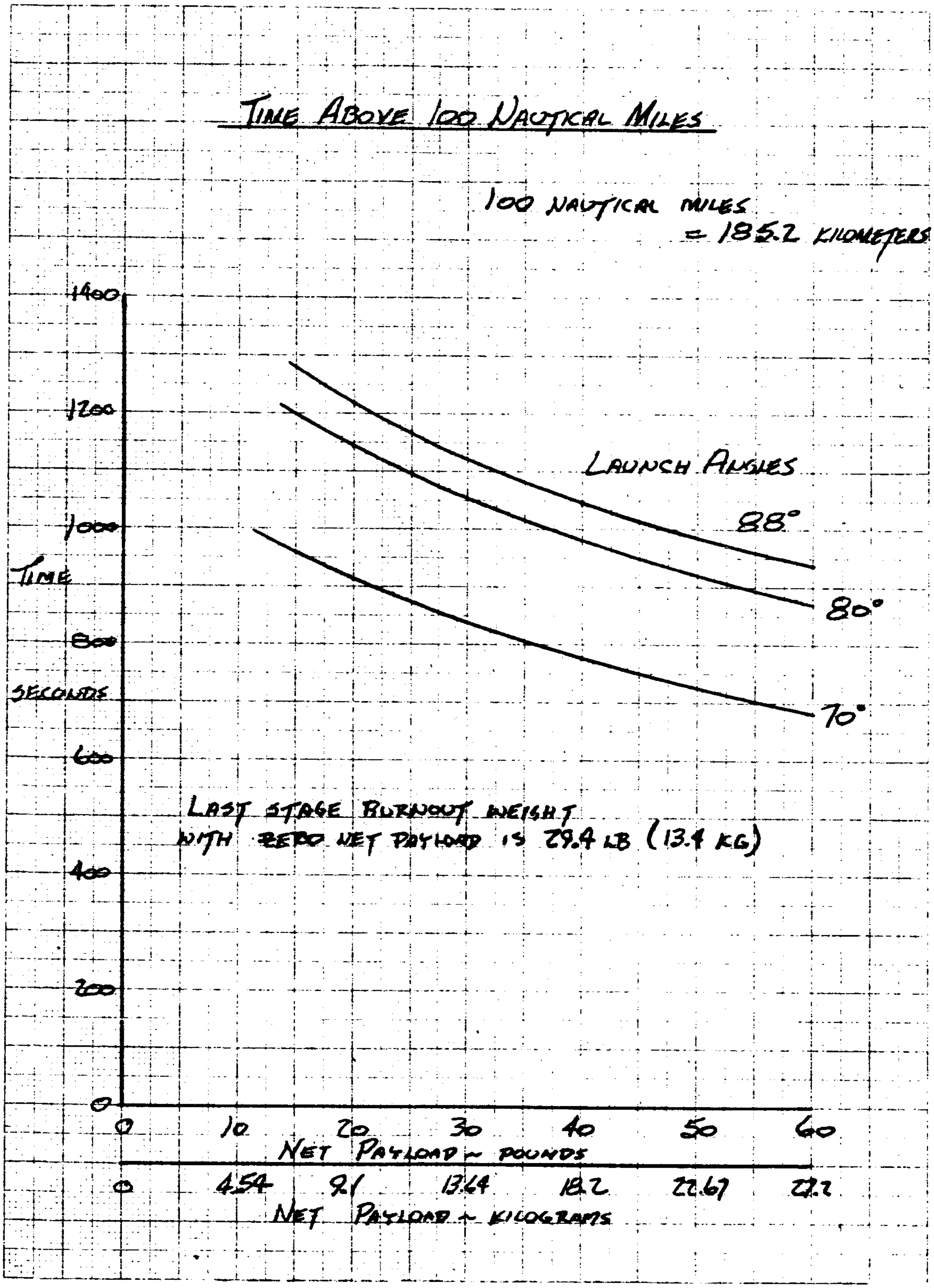



FIGURE 15 TIME ABOVE 100 NAUTICAL MILES

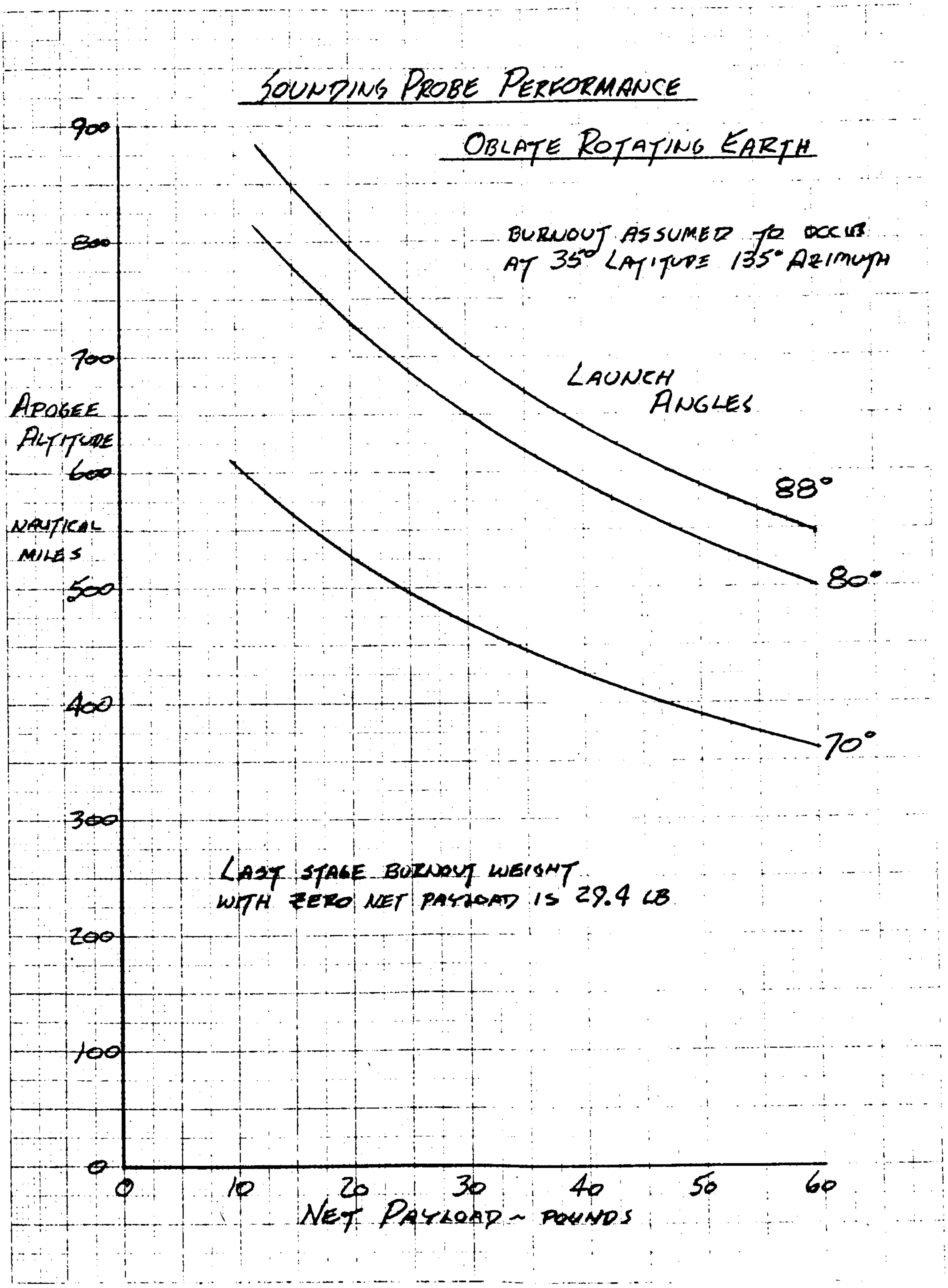



FIGURE 16 SOUNDING PROBE PERFORMANCE OBLATE ROTATING EARTH

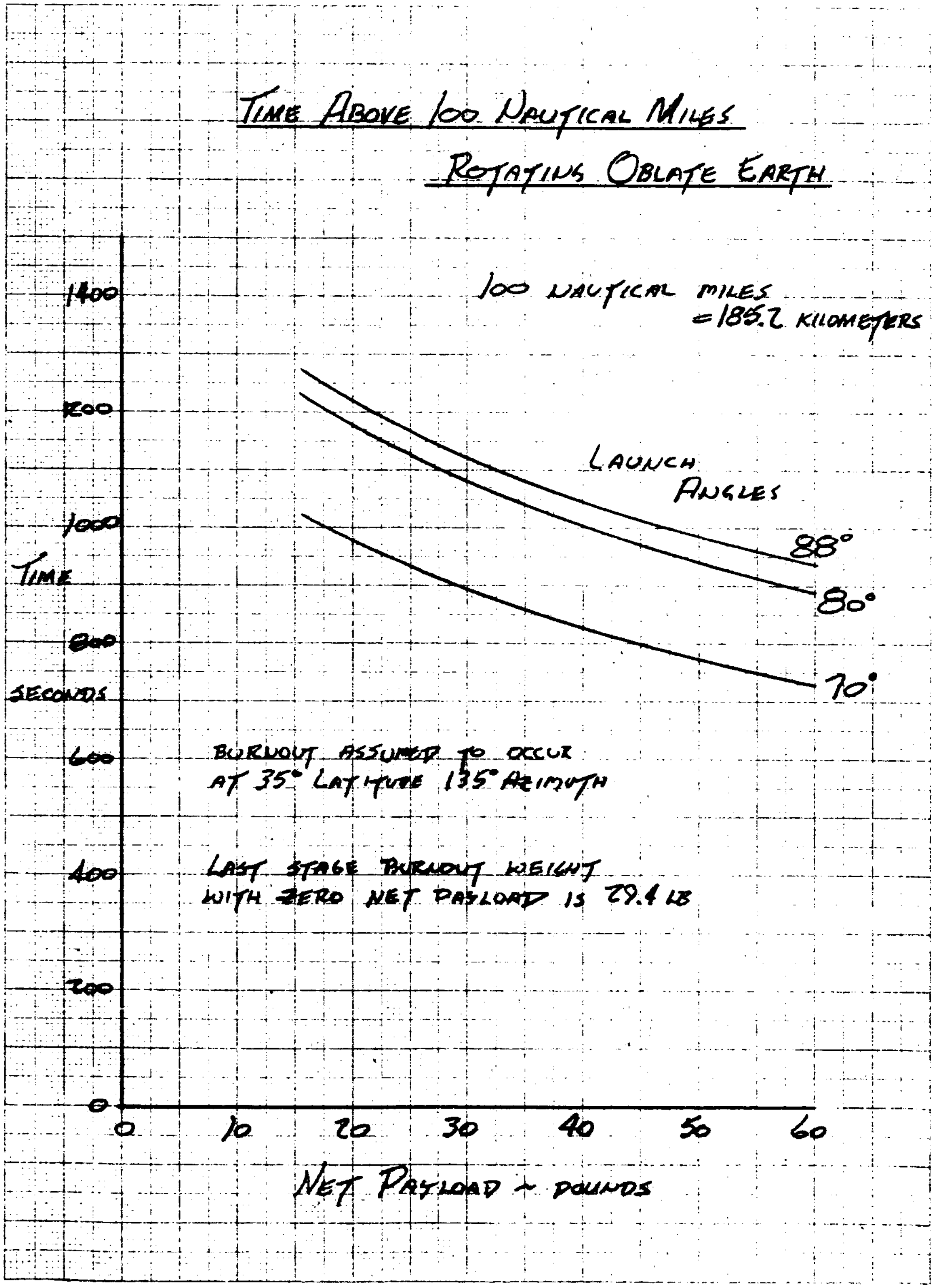



FIGURE 17 TIME ABOVE 100 NAUTICAL MILES ROTATING OBLATE EARTH

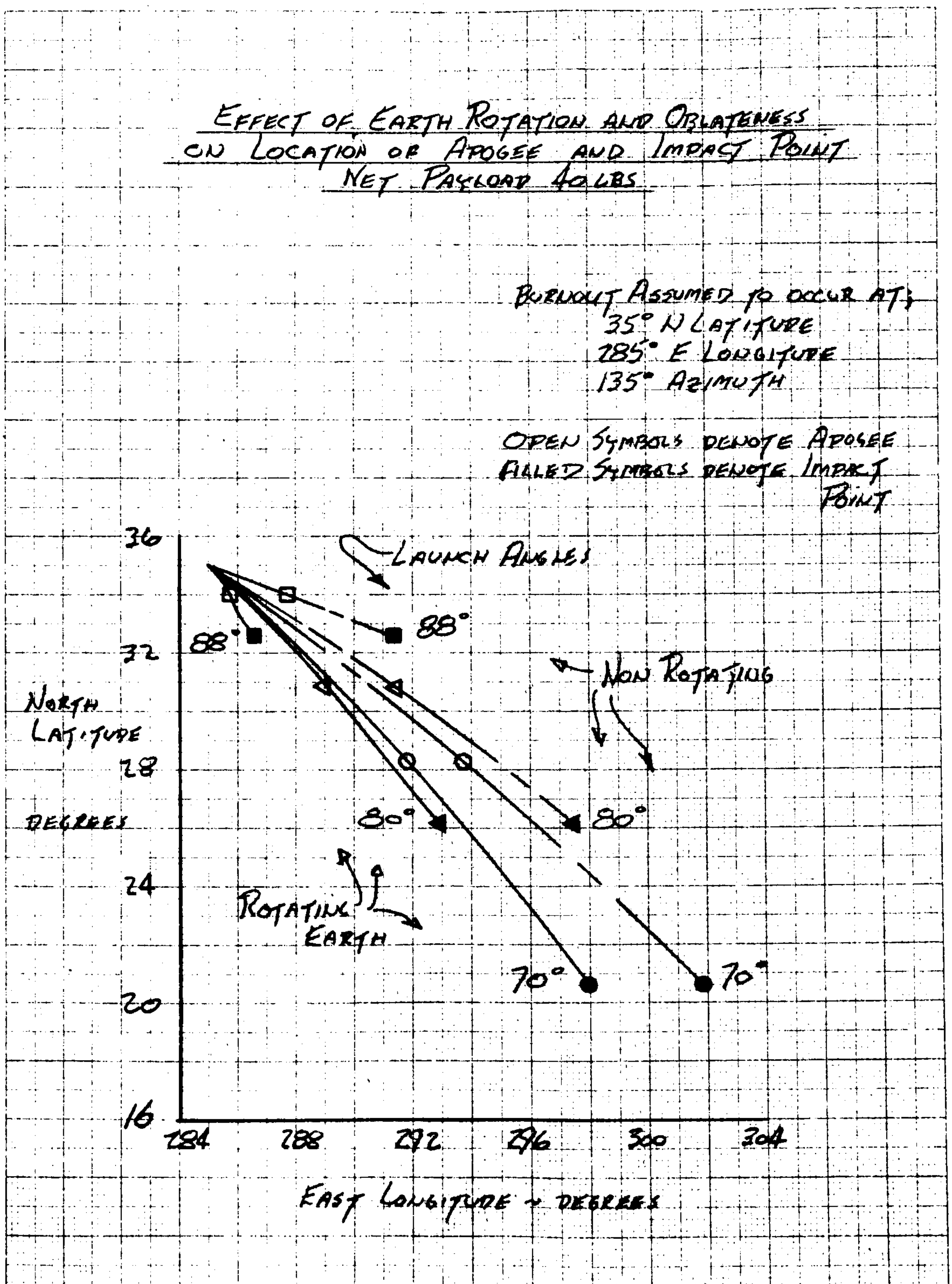



FIGURE 18 EFFECT OF EARTH ROTATION AND OBLATENESS ON LOCATION OF APOGEE AND IMPACT POINT NET PAYLOAD 40 LBS.

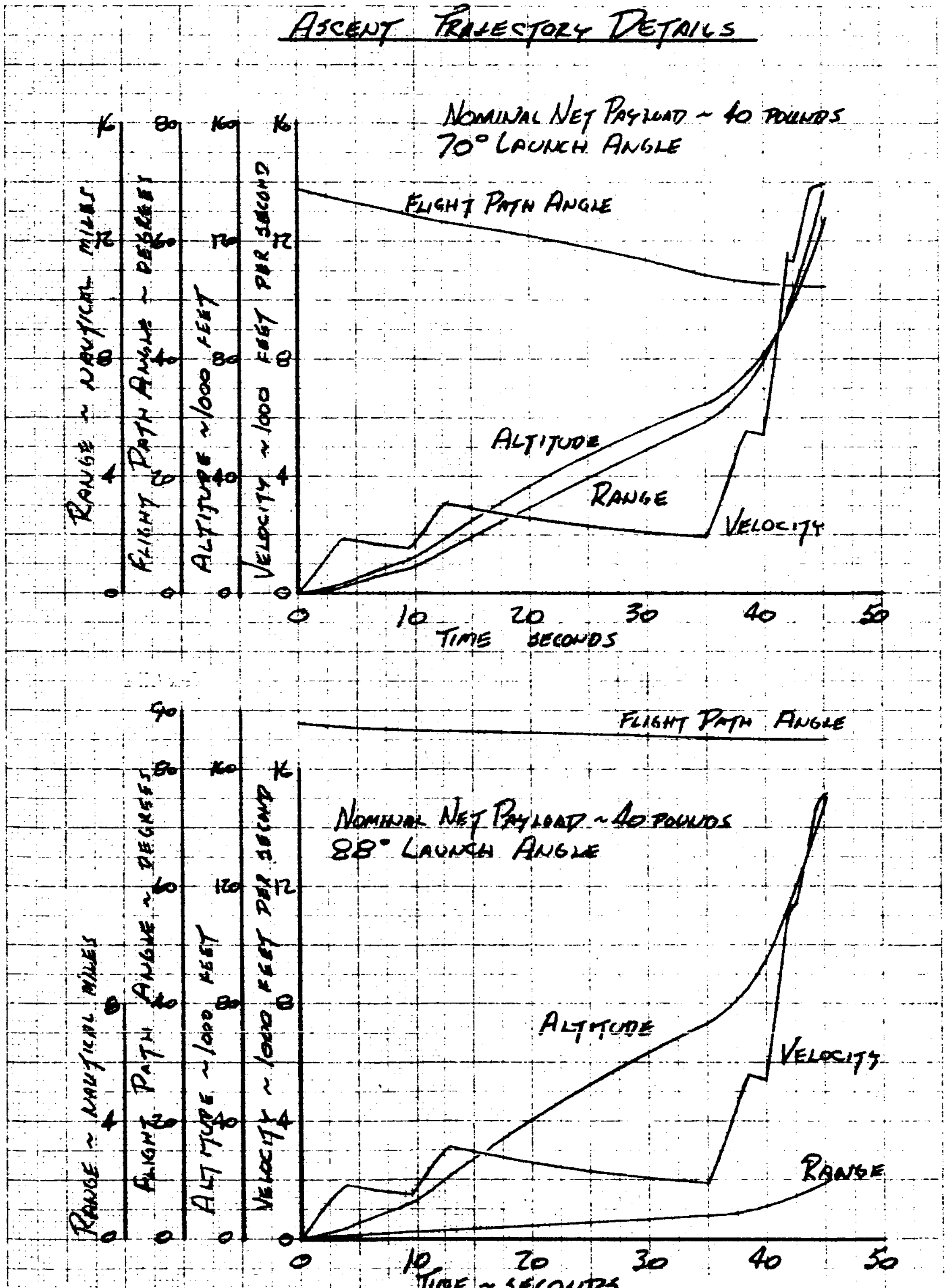



FIGURE 19 ASCENT TRAJECTORY DETAILS ($70^{o}$ and $80^{o}$ LAUNCH)

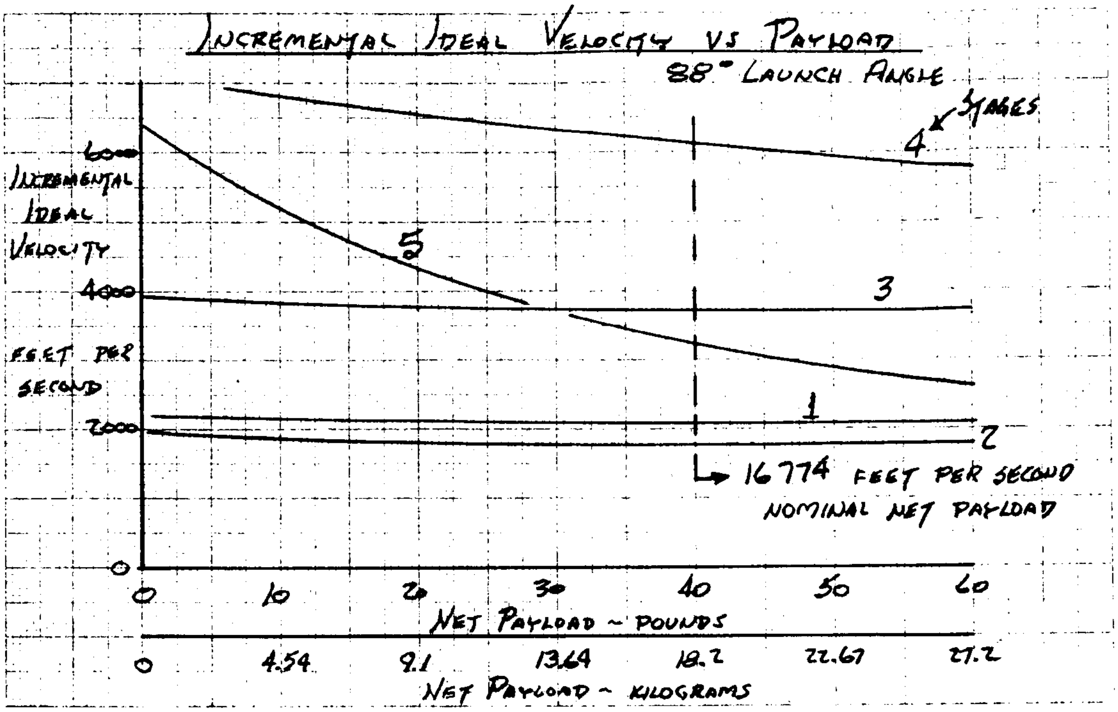



FIGURE 20 INCREMENTAL IDEAL VELOCITY Vs. PAYLOAD

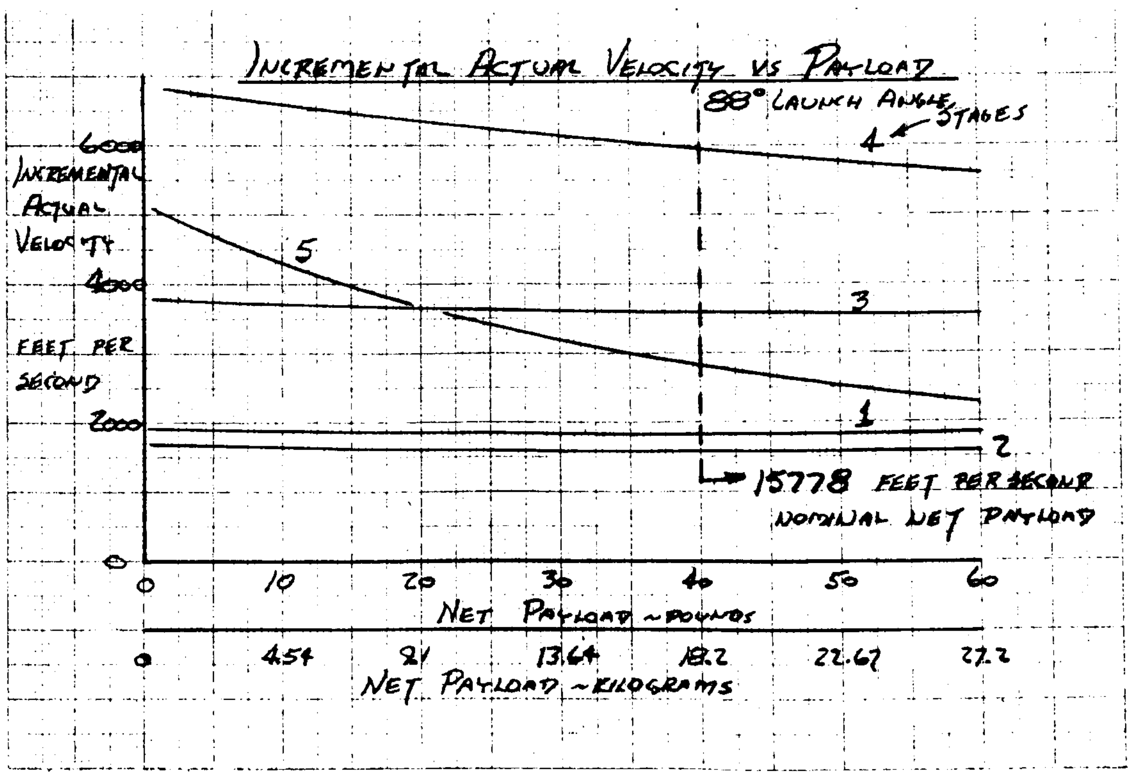



FIGURE 21 INCREMENTAL ACTUAL VELOCITY Vs. PAYLOAD

ACTUAL AND IDEAL
VELOCITY TABULATION

| LAUNCH ANGLE 88° | NOMINAL NET PAYLOAD 40 LBS. | | | |
|---|---|---|---|---|
| PHASE OF FLIGHT | IDEAL VELOCITY INCREMENT (FT/SEC) | VELOCITY LOST TO DRAG AND GRAVITY (FT/SEC) | COAST VELOCITY LOST (FT/SEC) | ACTUAL VELOCITY INCREMENT (FT/SEC) |
| STAGE 1 BOOST | 2085. | 231. | | 1854. |
| STAGE 2 COAST | | | 371. | -371. |
| STAGE 2 BOOST | 1799. | 221. | | 1578. |
| STAGE 3 COAST | | | 1175. | -1175. |
| STAGE 3 BOOST | 3735. | 136. | | 3599. |
| STAGE 3 COAST | | | 65. | -65. |
| STAGE 4 BOOST | 6130. | 174. | | 5956. |
| STAGE 5 BOOST | 3025. | 234. | | 2791. |
| TOTALS | 16774. | 996. | 1611. | 14167. |

FIGURE 22 ACTUAL AND IDEAL VELOCITY TABULATION

# BASIC DATA

## Weight

Detail Weight, Center of Gravity, and Inertia Data

The following detailed weight breakdown was used in all weight, c.g., and inertia calculations. Weight information was taken from Aerolab Drawing No. R12170. Motor inert weights and consumed weights were modified in order to present consistent data for all vehicles using the same motors. In addition, a nose cone weight of 10 pounds was assumed in order to present performance based on net payload.

Overall vehicle weight, c.g., and inertia values versus time for varying payload weights are presented in Figures 23 through 28.

| Item | Weight Pounds | C.G., In. from Ref. Sta. 0.0 | Local Roll Moment of Inertia, Slug-Feet$^2$ | Local Pitch or Yaw Moment of Inertia, Slug-Feet$^2$ |
|---|---|---|---|---|
| Fifth Step | | | | |
| Nose Cone | 10.0 | 27.0 | .0194 | .4588 |
| Motor Inert | 12.8 | 70.0 | .0227 | .3221 |
| Motor Shell & Nozzle Ext. | 6.6 | 75.0 | .0127 | .2558 |
| Step Total-Empty | 29.4 | 56.54 | | |
| Consumed Weight | 33.2 | 64.0 | .0358 | .4765 |
| Step Total-Loaded | 62.6 | 60.50 | | |
| Fourth Step | | | | |
| Adaptor | 7.3 | 94.2 | .0289 | .0173 |
| Motor Inert | 90.0 | 154.0 | .3944 | 21.24 |
| Skirt | 17.0 | 201.7 | .1321 | .2019 |
| Step Total-Empty | 114.3 | 157.28 | | |
| Consumed Weight | 268.0 | 142.0 | .7206 | 37.69 |
| Step Total-Loaded | 382.3 | 146.57 | | |

| Item | Weight Pounds | C.G., In. from Ref. Sta. 0.0 | Local Roll Moment of Inertia Slug-Feet$^2$ | Local Pitch or Yaw Moment of Inertia Slug-Feet$^2$ |
|---|---|---|---|---|
| Third Step | | | | |
| Adaptor | 21.6 | 211.5 | .2984 | .1608 |
| Motor Inert | 431.0 | 277.6 | 6.328 | 144.4 |
| Fins | 88.4 | 330.2 | 4.839 | 2.741 |
| Step Total-Empty | 541.0 | 283.56 | | |
| Consumed Weight | 764.0 | 269.8 | 5.279 | 145.7 |
| Step Total-Loaded | 1305.0 | 275.50 | | |
| Second Step | | | | |
| Adaptor | 32.4 | 479.8 | .1427 | .2099 |
| Motor Inert | 431.0 | 413.2 | 6.378 | 144.4 |
| Fins | 94.9 | 467.0 | 5.195 | 2.942 |
| Step Total-Empty | 558.3 | 426.21 | | |
| Consumed Weight | 764.0 | 405.4 | 5.279 | 145.7 |
| Step Total-Loaded | 1322.3 | 414.19 | | |
| First Step | | | | |
| Adaptor | 46.0 | 491.4 | .9385 | .5208 |
| Fins | 252.0 | 666.3 | 32.59 | 22.90 |
| Inert | 1692.0 | 595.0 | 44.93 | 1204. |
| Step Total-Empty | 1990.0 | 601.63 | | |
| Consumed Weight | 2180. | 569.5 | 28.95 | 873.9 |
| Step Total-Loaded | 4170.0 | 584.84 | | |
| Launch (no payload) | 7242.2 | 470.27 | | |

Payloads:

| | | | | |
|---|---|---|---|---|
| Minimum | 20.0 | 27.0 | .0194 | .7399 |
| Nominal | 40.0 | 27.0 | .0388 | 1.475 |
| Maximum | 60.0 | 27.0 | .0583 | 2.215 |

Notes

(1) Local roll moment of inertia of all fins and launch fittings is about vehicle centerline
(2) Sta. 0.0 is tip of nose cone of 688.4" long vehicle (assuming nose cone shape as shown in NASA Memo 3-6-59L)
(3) Weight data from Aerolab Drawing R12170.
(4) Motor data from latest information from motor manufacturers.
(5) All other data calculated.

Aerodynamics

The basic aerodynamic data for zero angle of attack are shown in Figure 29. It is probable that the center of pressure will move forward slightly as the angle of attack is increased.

Drag will increase moderately with angle of attack. The roll rate and pitch frequency time histories are shown in Figure 31. For this vehicle, the philosophy is to keep the roll rate lower than the natural frequency for the first two stages. During third stage burning the roll rate is increased to a level about twice the natural frequency. Because of the rapid change in the roll rate to pitch frequency ratio, the magnitude of the oscillation probably does not build up very high. In a case such as this, the drag increment resulting from pitch-roll coupling is probably insignificant.

Propulsion

Since the rocket motor ballistic data for some of the motors used on the 18 vehicles of this sounding rocket study series are classified, all of the ballistic data, both classified and unclassified, were consolidated into report no. AST/E1R-13336 so that all of the individual vehicle reports would remain unclassified.

The classification of the motors used on the Jason vehicle is:

| MOTOR | CLASSIFICATION |
|---|---|
| Honest John | Confidential |
| Nike-Ajax Booster | Confidential |
| Recruit | Confidential |
| T-55 | Confidential |

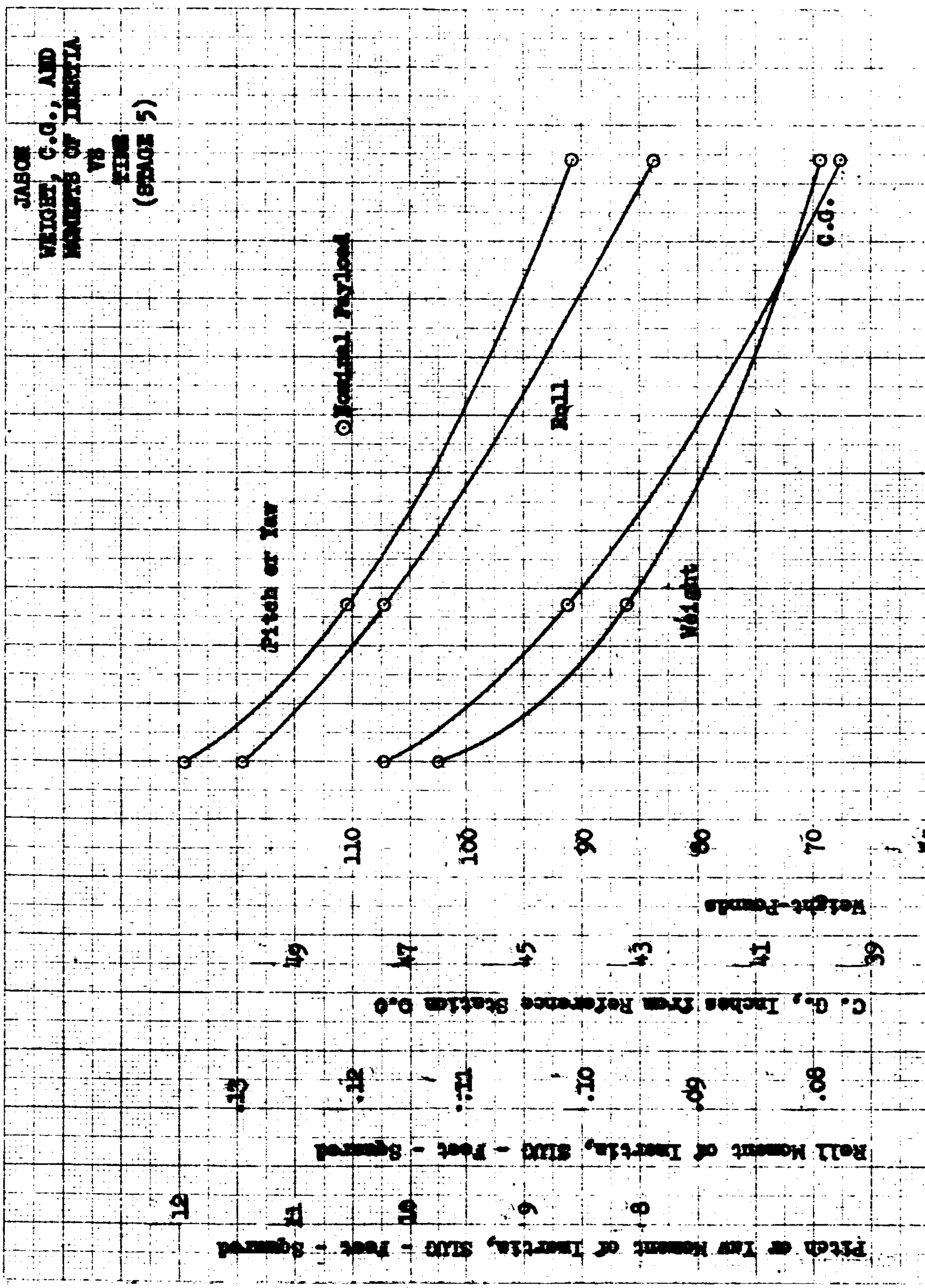



FIGURE 23 WEIGHTS, CENTER OF GRAVITY AND MOMENTS OF INERTIA Vs. TIME (STAGE 5)

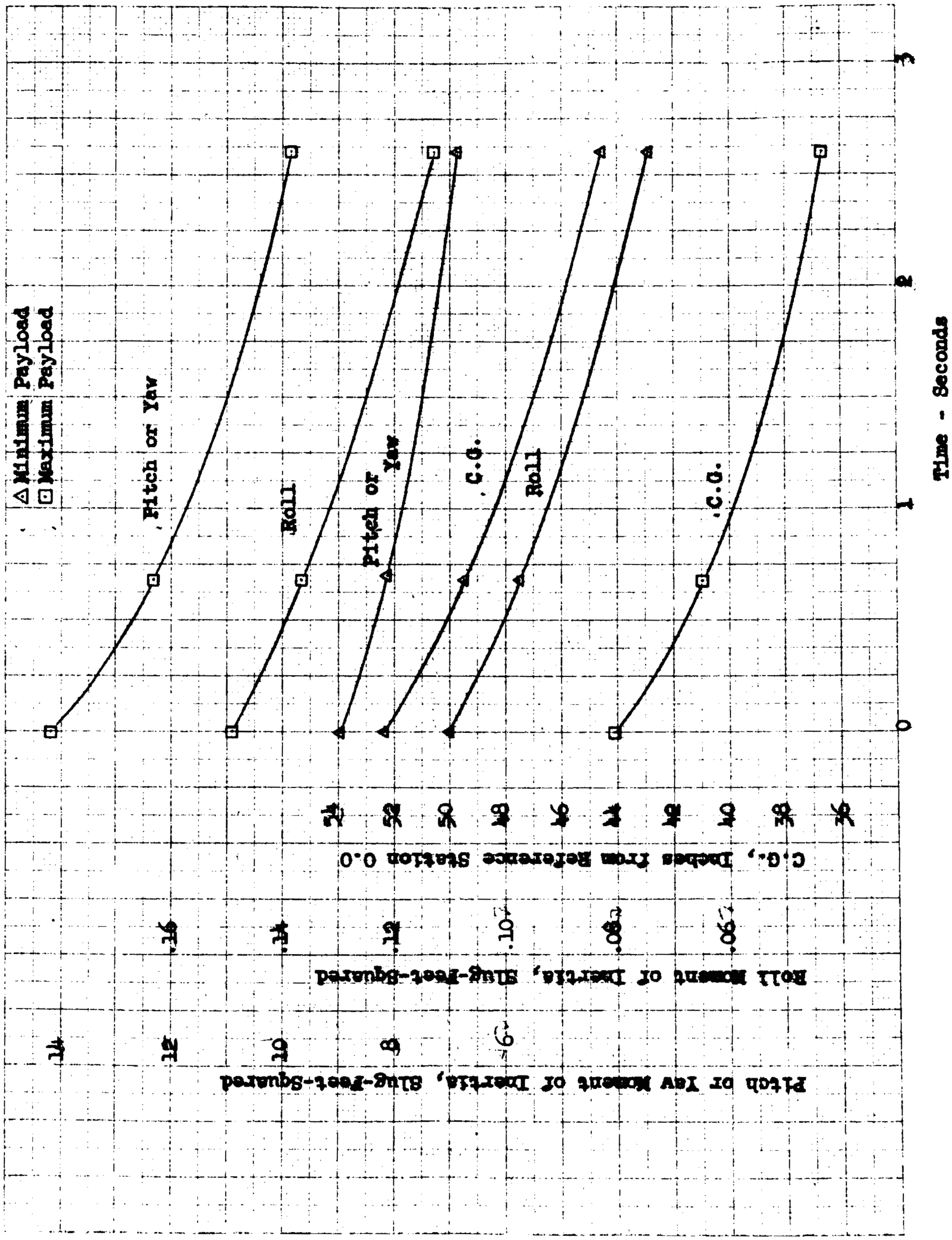



FIGURE 24 CENTER OF GRAVITY AND MOMENTS OF INERTIA Vs. TIME (STAGE 5)

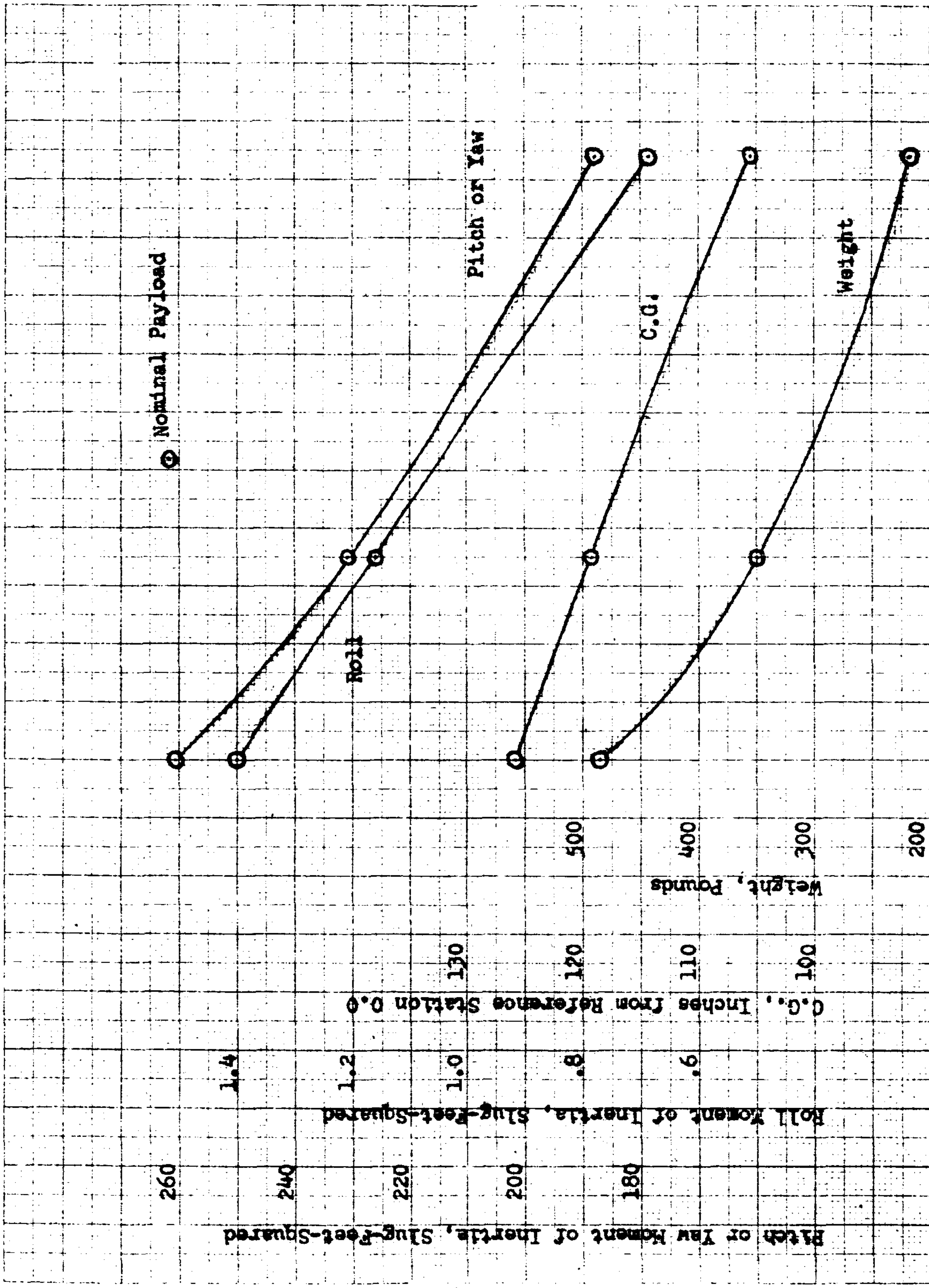



FIGURE 25 WEIGHT, CENTER OF GRAVITY AND MOMENTS OF INERTIA Vs. TIME (STAGE 4)

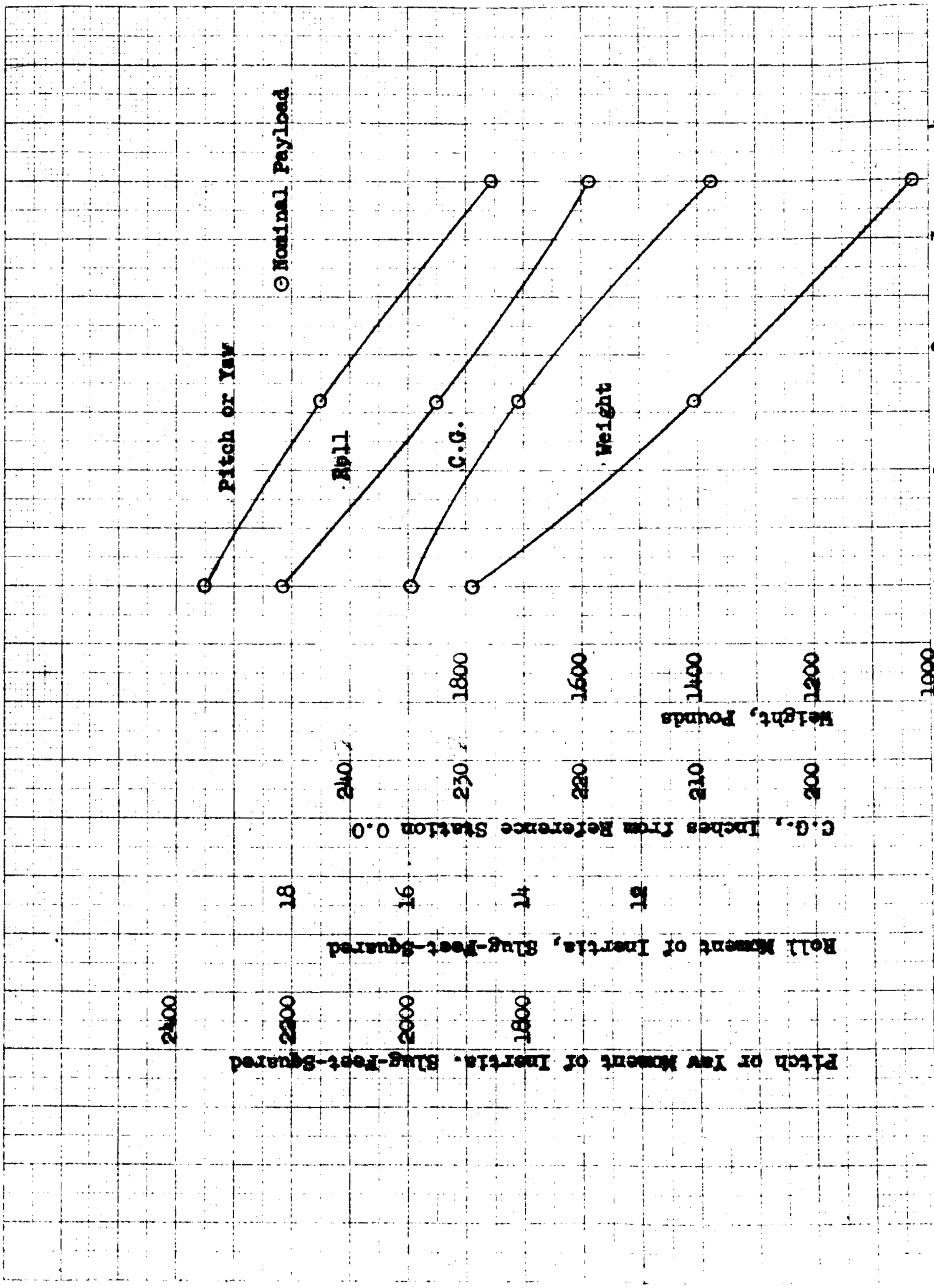



FIGURE 26 WEIGHT, CENTER OF GRAVITY AND MOMENTS OF INERTIA Vs. TIME (STAGE 3)

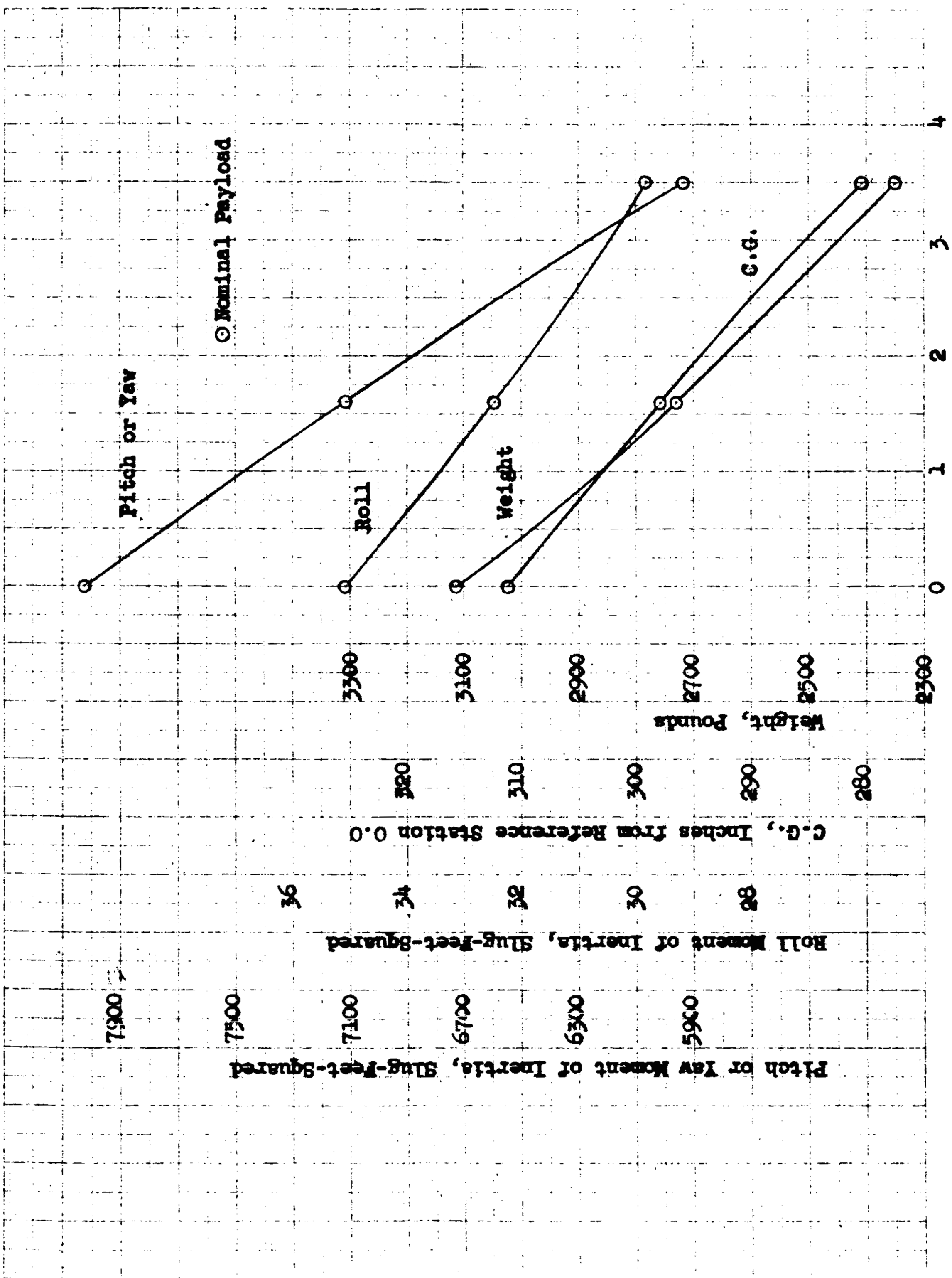



FIGURE 27 WEIGHT, CENTER OF GRAVITY AND MOMENTS OF INERTIA Vs. TIME (STAGE 2)

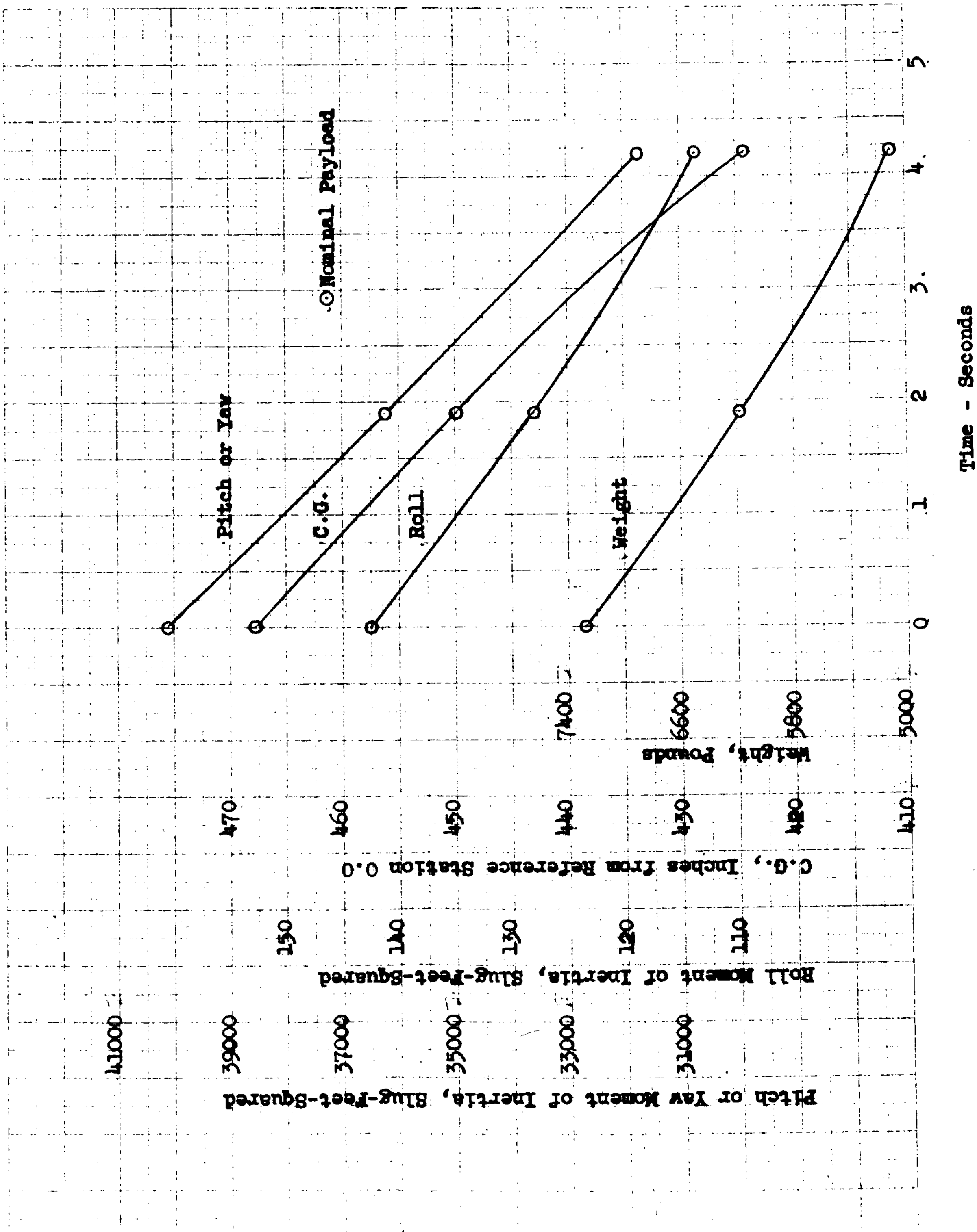



FIGURE 28 WEIGHT, CENTER OF GRAVITY AND MOMENTS OF INERTIA Vs. TIME (STAGE 1)

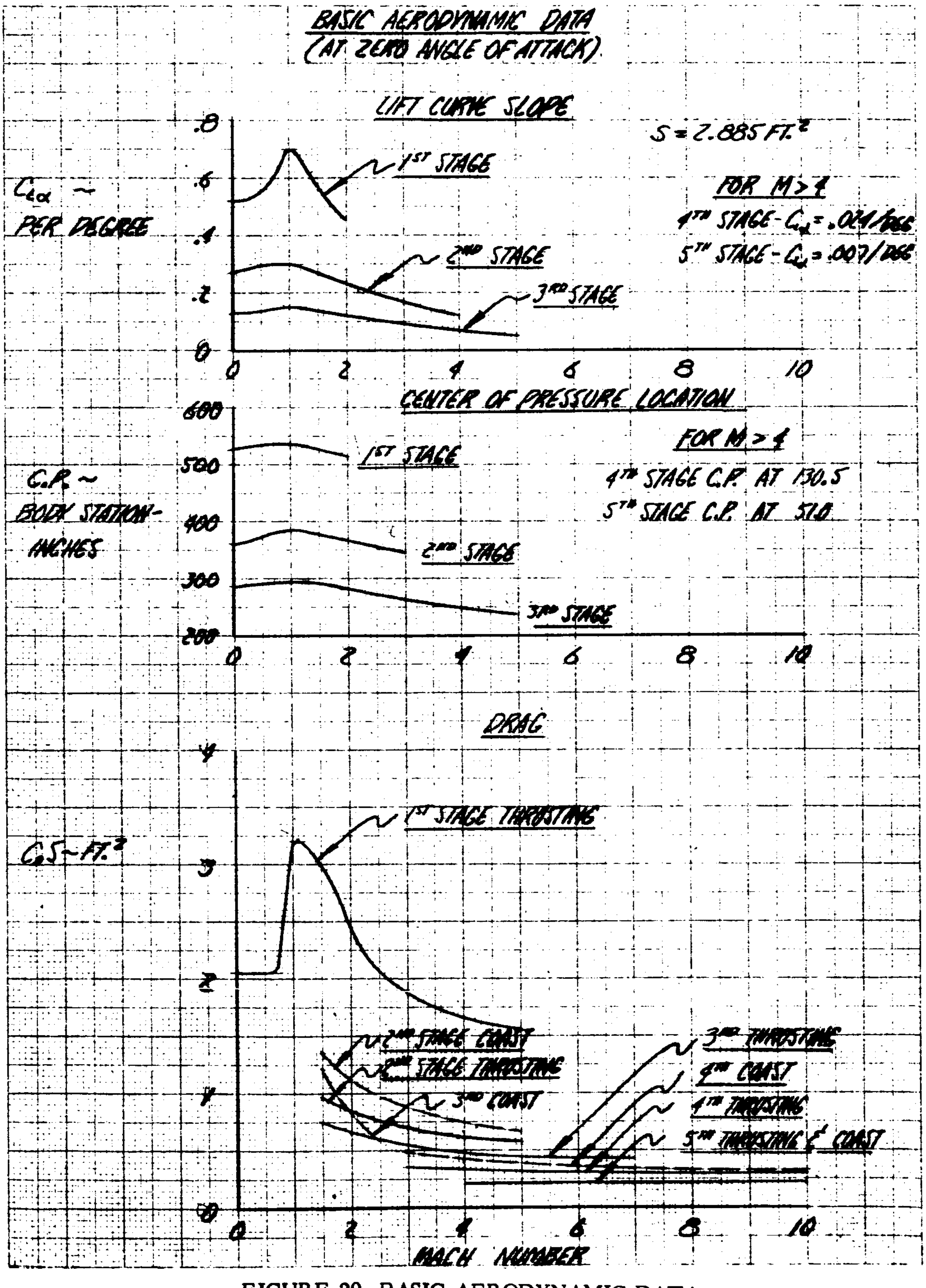



FIGURE 29 BASIC AERODYNAMIC DATA

# ENVIRONMENT

## Axial Acceleration

The axial acceleration time history of the JASON at a launch angle of 88° is shown on Figure 30.

## Roll Rate

The roll rate time history is shown in Figure 31. Tangential and centripetal accelerations acting on the payload in this case will probably be a minor consideration. The roll rate shown will produce pitch-roll coupling during third stage burning, but this will probably be a minor consideration.

## Structural Dynamic Analysis

The flexible vehicle structure is subjected to a number of loading environments which produce significant dynamic responses. The load inputs occur from ground handling, launch, atmospheric disturbances, control commands, stage separation, and structural and thrust misalignments. The spin stabilized vehicle experiences additional loading phenomena arising from dynamic coupling.

Atmospheric disturbances in the form of winds and gusts require an extensive analysis for structural loads determination. This involves a trajectory analysis of the flexible vehicle, taking into consideration time variations in weight and aerodynamic load distributions. Atmospheric winds and gusts are defined statistically so that ultimately the analysis produces a missile loading criteria in terms of probability of structural failure.

The weight distribution may be obtained from available information,but other information necessary to determine the structural dynamic characteristics of the JASON vehicle have not been received.

## Vibration

In order to obtain the vibration environment in the payload compartment, it is necessary to know the vibration characteristics of the sources, such as rocket motor(s), aerodynamic boundary layer noise, and launch noise.

Payload base input vibrations normally would be expressed as a function of vibration amplitude versus frequency for significant flight times, and with the characteristics of the vibration (i.e., sinusoidal, random, mixed) indicated. The resulting payload vibration environment will depend upon the structural dynamic response characteristics of the payload itself, in addition to the payload base input vibration environment.

No indications were found in the data available of any payload compartment vibration instrumentation provisions having been made for the Jason vehicle.

## Temperature

### External Temperature

To determine the temperature effects on this vehicle it was necessary to select a given trajectory and specific components to be investigated. The 70° launch angle and nominal payload were selected as a limiting condition which would emphasize possible mission restrictions that result from skin heating. A vehicle which has been used at launch angles above 80° might be inadequate for a 70° launch. This is shown to be the case for the Jason vehicle from the temperature curves of Figure 32.

The components investigated include the nose cone and fin stagnation areas, the nose cone fairing in the payload area, and the fin panels as shown in Figure 32. While these areas normally experience maximum heating, this does not imply that other areas on the vehicle, such as rocket cases, do not require investigation for a particular mission.

Skin gages shown in Figure 32 were obtained from the manufacturers and from information available at NASA for the fin leading edge wrapped configuration.

The physical properties for the transient temperature analysis, using digital computer methods, are shown below:

| Material | Titanium | Composite Fin L. E. | Magnesium |
|---|---|---|---|
| Density ($lb/ft^3$): | 276. | 106. | 106. |
| Specific Heat (Btu/Lb-°F): | .137 | .33 | .28 |
| Emissivity: | .35 | .6 | .6 |

To properly evaluate the structural reliability of any vehicle, the load-temperature relationship with respect to time must be considered. This relationship cannot be adequately defined until a specific mission requirement has been selected. For example, a component may experience severe reduction in strength allowables due to temperature, but if the elevated temperature occurs at times when the load is negligible, the condition may be acceptable.

The titanium nose cone and fairing experience strength reductions due to temperature to approximately 25% and 50%, respectively, of their room temperature properties at 50 seconds after launch. The loading and margins of safety at this time should therefore be investigated. The third stage magnesium fins would be inadequate in the temperature ranges shown for this mission. Either some form of thermal protection or use of a different material for these fins is indicated for the 70° launch. Higher launch angles would reduce these temperatures to a more acceptable value.

Internal Heating of Payload Compartment

The payload compartment temperature while the vehicle is on the launch pad is a function of the ambient temperature, location of the launch pad, time on the launch pad, and the heat output of the payload.

To determine the payload compartment temperature while the vehicle is on the launch pad, an average payload with an area-weight ratio of 0.1 $ft^2$/lb. was considered. The compartment walls were assumed to be gold-coated (due to the low emissivity of gold) and the compartment subjected to an ambient temperature of 100 F. The compartment temperatures were calculated for payload power outputs of 10, 100, and 200 watts which correspond to payload power densities of 0.1, 1.0 and 2.0 watts/lb., respectively. The compartment temperatures were calculated considering convection, radiation, and storage of heat by the payload. Considering these conditions, payloads with a power density of 2.0 watts/lb. or above will require additional cooling to hold the compartment temperature to 150°F or below, if they remain on the launch pad with power from one to two hours prior to launch (which is generally not normal procedure). The usual pre-launch "power on" condition is of relatively short duration and therefore pre-launch temperature is not normally a problem. The maximum compartment temperature limit for most electronic equipment is 150°F. Additional cooling of the payload compartment, if necessary, may be accomplished by forced ventilation, cooling to a subcooled state prior to launch, and by the addition of heat sinks to the payload.

The heating of the payload compartment after launch is a function of the compartment temperature prior to launch, vehicle flight path, duration of flight, heat output of the payload, and compartment configuration.

To determine the payload compartment temperature after launch, payloads of the same magnitude as above were considered. A nominal atmospheric trajectory was used to determine the effects of aerodynamic heating on the compartment. Payloads with a power density of 1.0 watts/lb. or above and a flight time in excess of 30 minutes will require additional cooling prior to launch to hold the compartment temperature to 150°F or below based on a launch temperature of 100 F. Payloads with a power density of 0.1 watt/lb. or below will require additional cooling if the flight time exceeds one hour based on the aforementioned launch temperature. The flight time of the Jason vehicle is approximately 20 minutes. However, the environment of each payload should be further analyzed with respect to the conditions stated in paragraph 3 prior to establishing the payload cooling requirements.

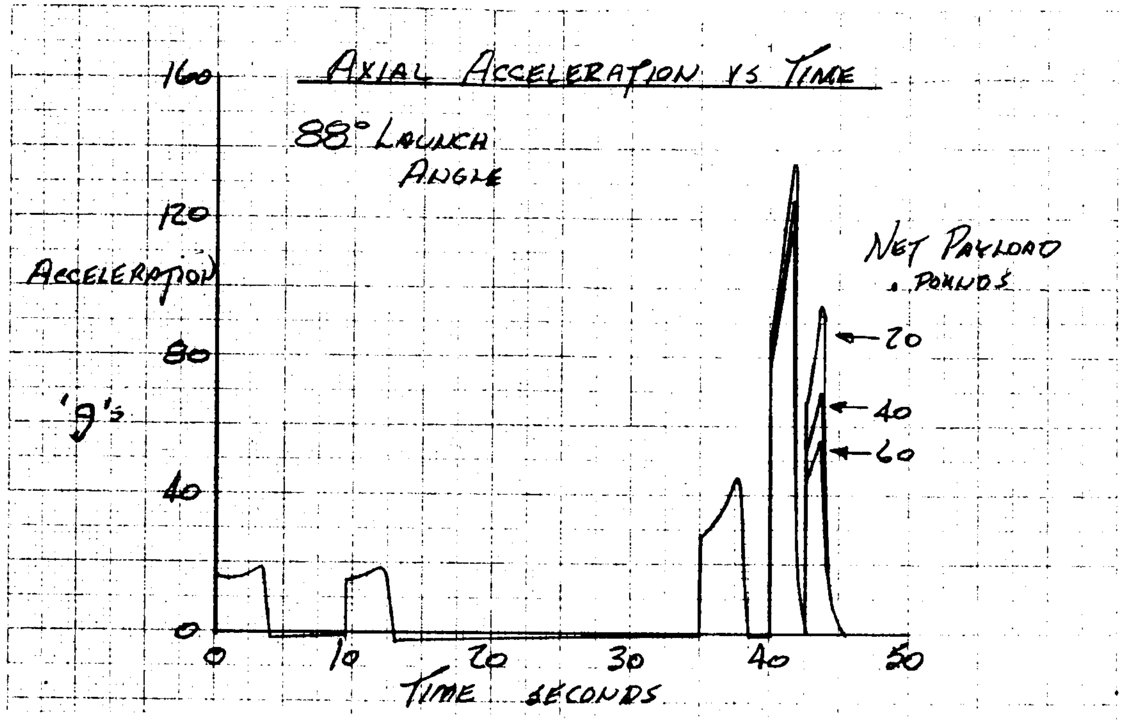



FIGURE 30 AXIAL ACCELERATION Vs. TIME

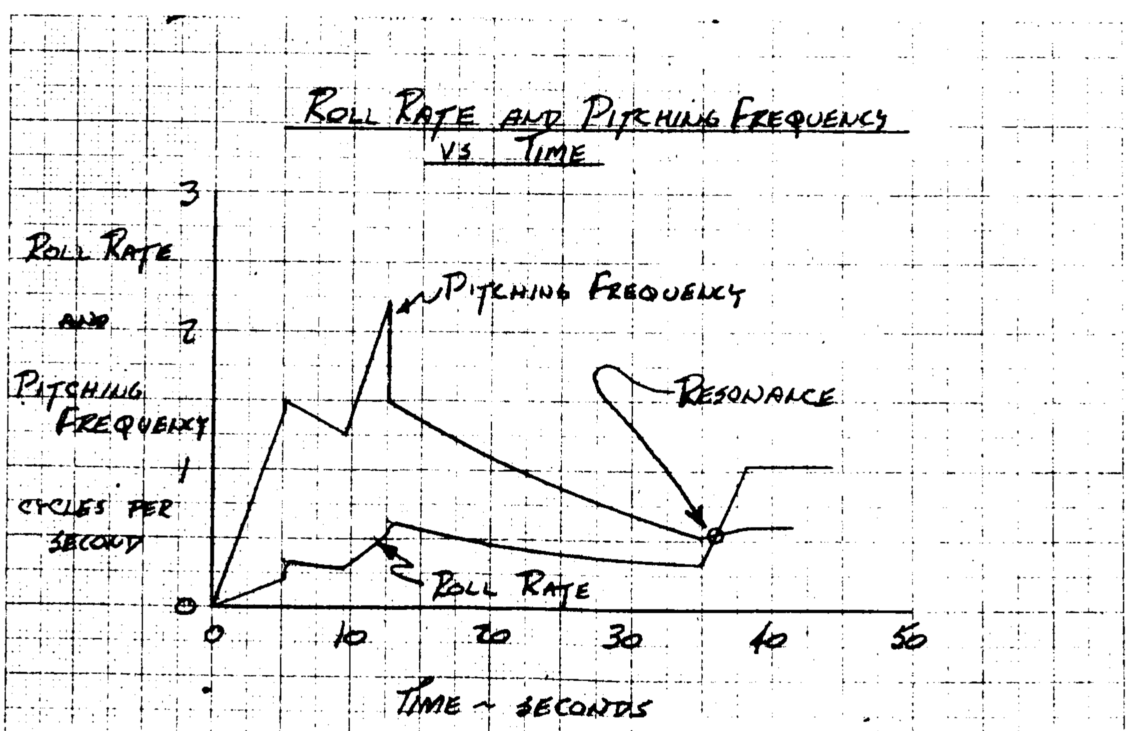



FIGURE 31 ROLL RATE AND PITCHING FREQUENCY Vs. TIME

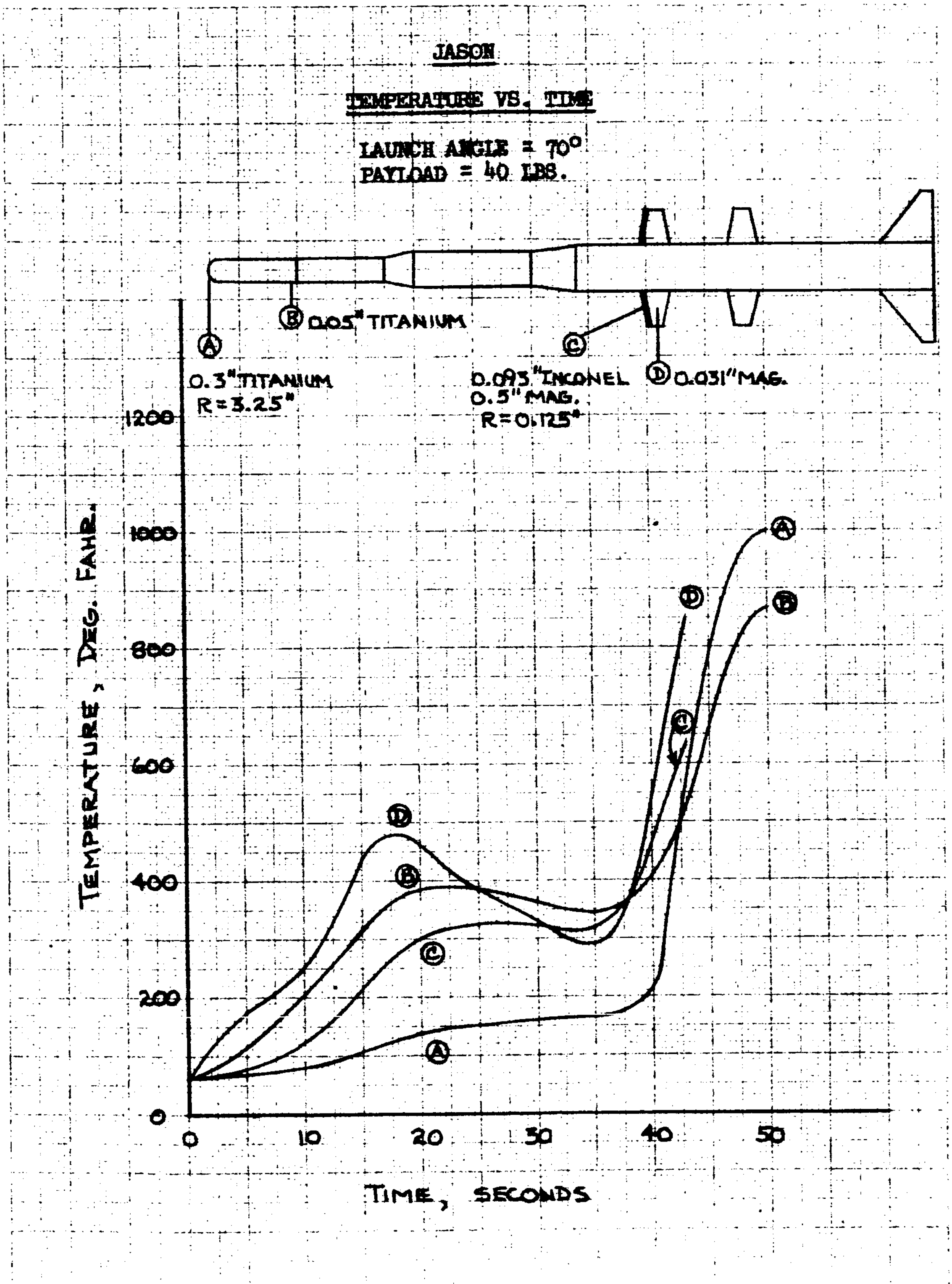



FIGURE 32 TEMPERATURE Vs. TIME

# OPERATIONAL FACTORS

## Ground Support Equipment

### Mechanical Support Equipment

The Jason vehicle is launched from a modified Honest John I-beam launcher (Figures 33 & 34) or a modified tubular Sergeant Launcher (Figure 35). Contractor supplied handling equipment required are handling slings for the motor, holding fixture and miscellaneous tools. Government furnished equipment includes three Air Log Dollies, work stands, motor storage facility, air compressor for boom hoisting motor, and a fork lift truck for azimuth adjustment and general use.

### Electrical Support Equipment

First step ignition is accomplished with a 10V potential at approximately 9 amperes, and second stage delay ignitor is actuated at launch with 5 volts, 6 amperes. Equipment required includes an Allegany Instrument Company Model 101-5A squib tester, two battery chargers and load testers for stage ignition batteries, calibration equipment for the timer used for third and fourth stage ignition batteries, and calibration equipment for the timer used for third and fourth stage ignition.

## Instrumentation

The majority of sounding rocket vehicle flights to date did not require vehicle instrumentation since they had fixed fins, were unguided, and range safety requirements were not critical. Instrumentation may be desired on future flights to supplement payload data, verify trajectory characteristics, record staging sequences, monitor critical environmental conditions, and assure command destruct capability. Generally, the the information necessary to evaluate the instrumentation required would be: type of measurement desired, range, accuracy, frequency response, and resolution. Consideration must also be given to environmental requirement, the type of ground data gathering equipment already available, and duration of operation.

Instrumentation for a typical test vehicle (Reference 1) consisted of three accelerometers (x-, y-, and z- directions). Data from these instruments were transmitted and ground recorded continuously by an FM telemetering and ground receiving station. The transmitter in this test vehicle had a nominal power output of 8 to 10 watts.

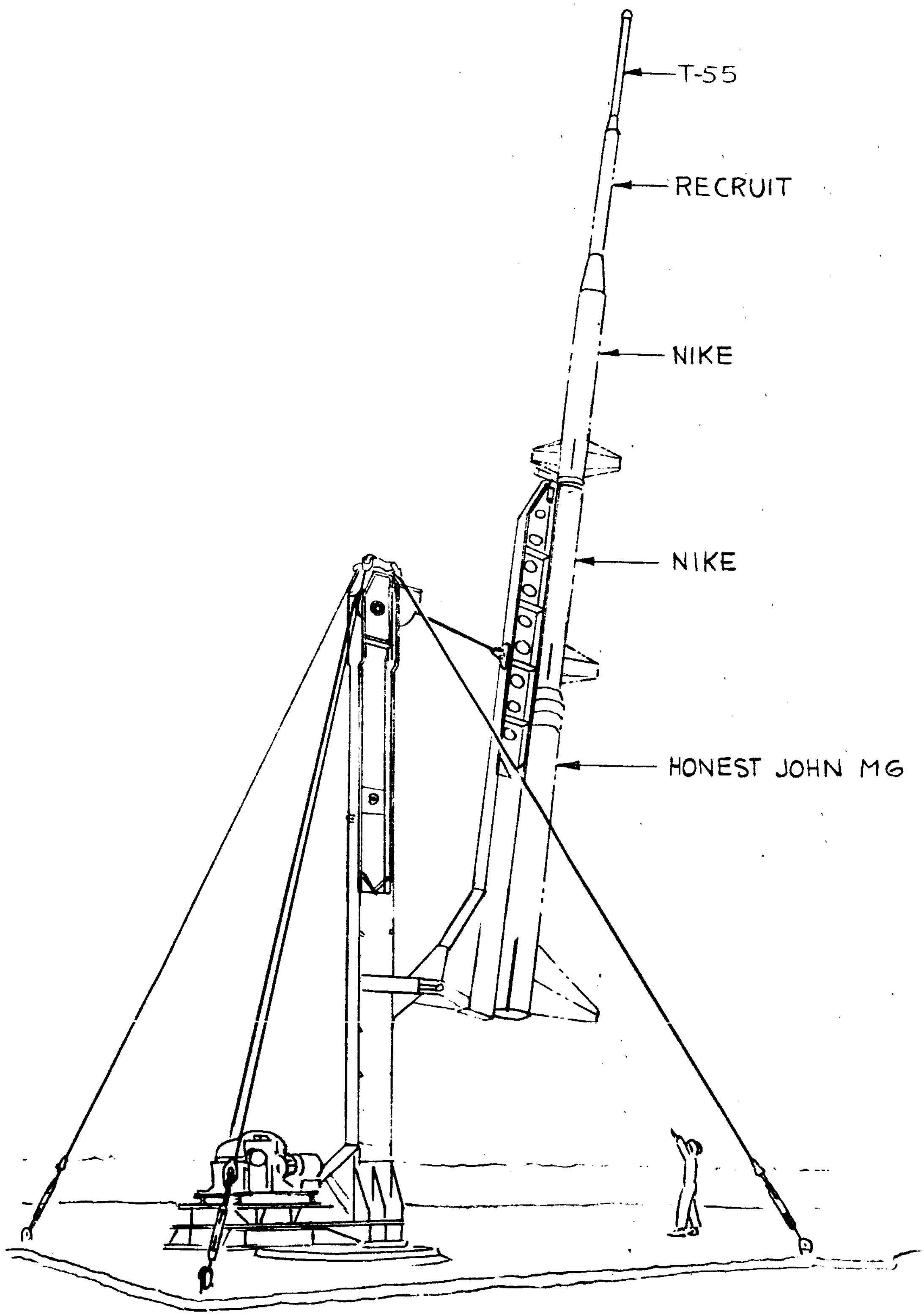



FIGURE 33 JASON IN LAUNCH POSITION ON MODIFIED HONEST JOHN LAUNCHER

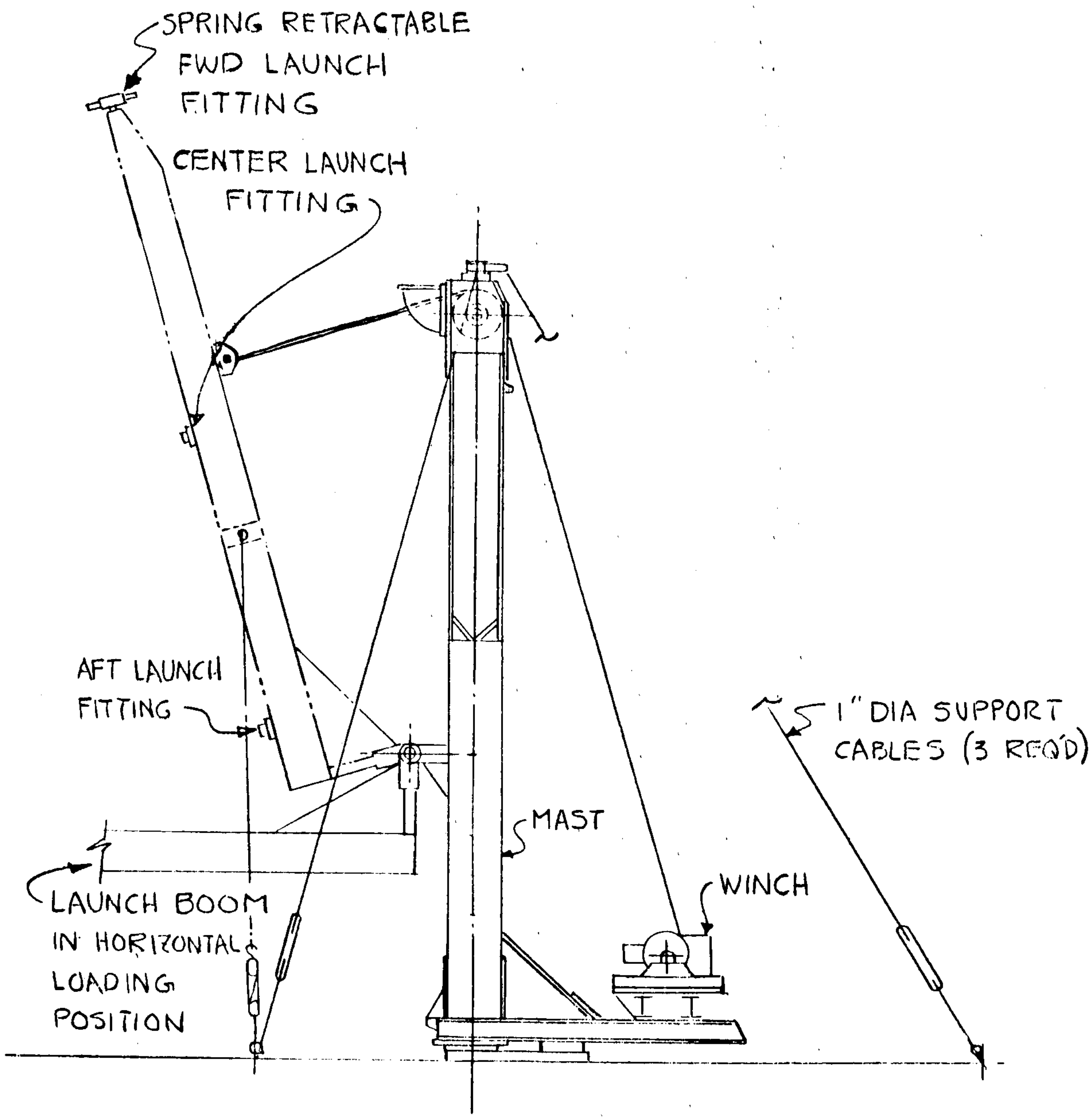



FIGURE 34 MODIFIED HONEST JOHN I BEAM LAUNCHER

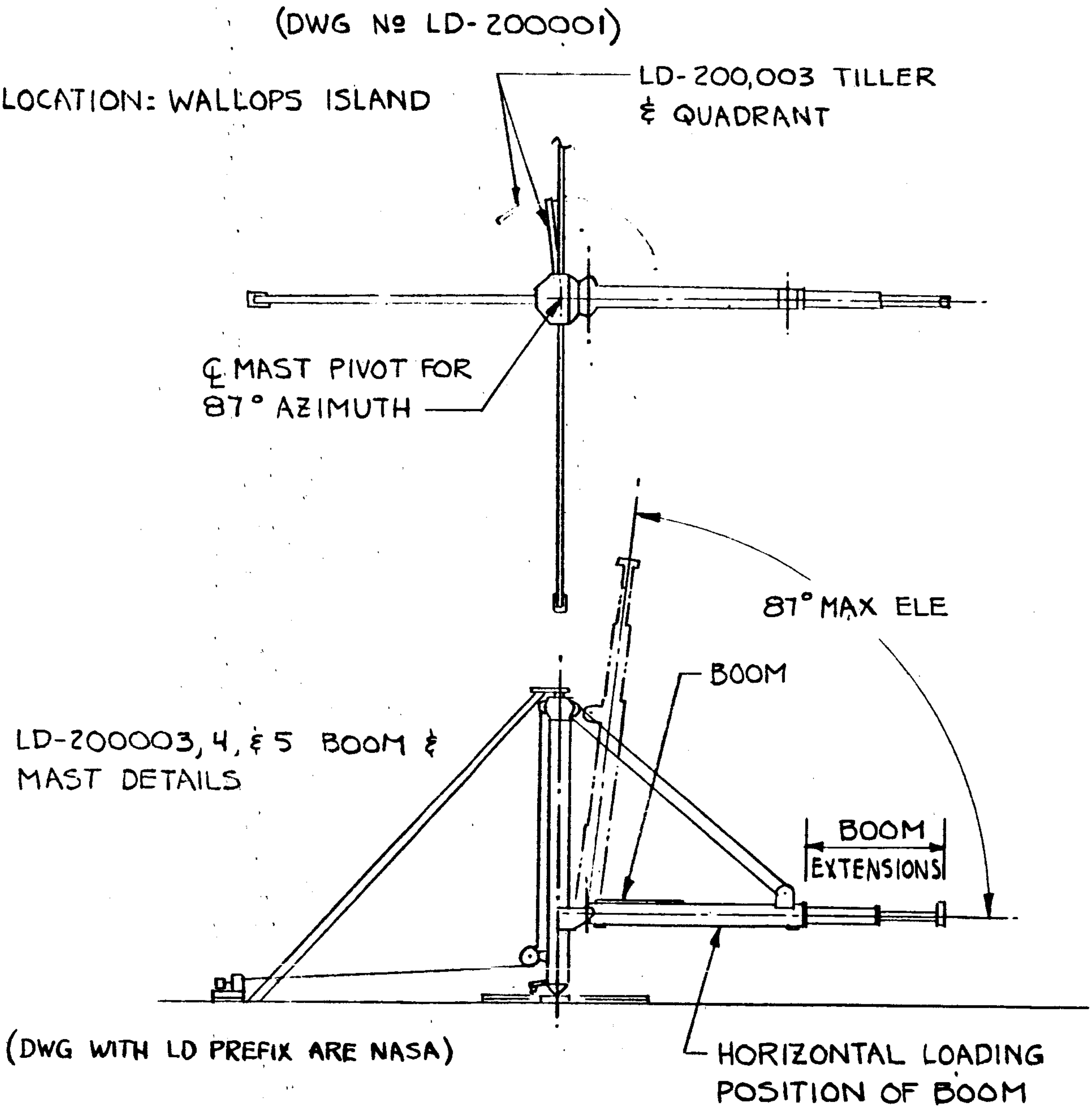



FIGURE 35 MODIFIED TUBULAR SERGEANT LAUNCHER

## NOMENCLATURE

| Symbol | Definition | Units |
|---|---|---|
| $C_{L_a}$ | Lift curve slope | per degree |
| $C_D$ | Drag coefficient | |
| c.g. | Center of gravity from reference datum | in. |
| C.P. | Center of pressure from reference datum | in. |
| $g$ | Gravitational acceleration | $ft/sec^2$ |
| $g_o$ | Gravitational acceleration at earth's surface* | $ft/sec^2$ |
| $g_s$ | Standard or normal gravitational acceleration | $32.174\ ft/sec^2$ |
| G | Vibrational acceleration | $ft/sec^2$ |
| $I_{TOT}$ | Total impulse | lb-sec |
| $(I_{sp})_{AVG}$ | Average specific impulse, $\frac{I_{TOT}}{w_c}$ | sec |
| $w_c$ | Total consumed weight | lb |
| $w_p$ | Weight of propellant | lb |
| $w_o$ | Weight of stage | lb |
| $R_o$ | Earth radius* | ft |
| S | Aerodynamic reference area | $ft^2$ |
| $\Delta V_{ID}$ | Ideal incremental velocity | ft/sec |
| $\alpha$ | Angle of attack | degrees |
| $\mu$ | Mass ratio, $\frac{w_o}{w_o - w_c}$ | |

* Where $g_o$ and $R_o$ represent conditions at a geodetic latitude of 35° on the International Ellipsoid of Reference:

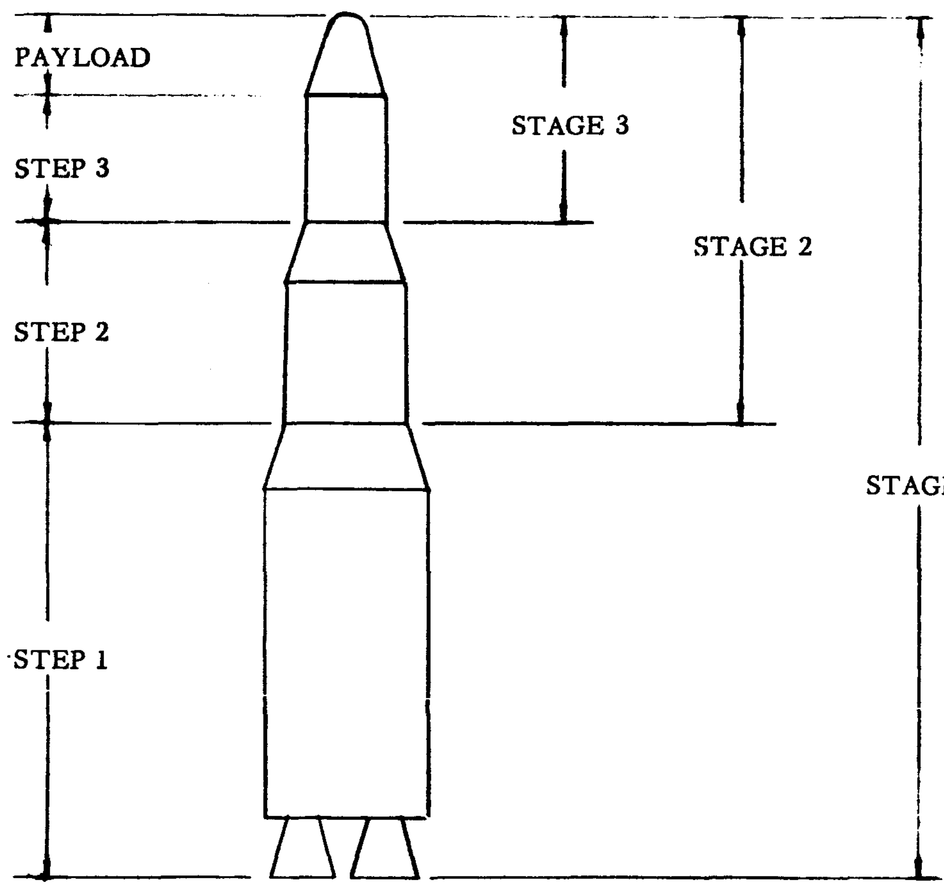
$$g_o = 32.14389\ ft/sec^2 = 9.797459\ m/sec^2$$

$$R_o = 20,903,307\ \text{feet} = 6371.328\ \text{kilometers}$$

## PAYLOAD DEFINITIONS

NET PAYLOAD: All weight not essential to the flying of the vehicle when the payload carrying stage is thrusting, but not including weight which is essential to the operation of a previous stage and which happens to remain attached to the payload carrying stage during its thrusting period.

## VEHICLE STAGING DEFINITION




"Stage" is the preferred nomenclature when referring to system operation. "Step" is the preferred nomenclature when referring to the precise location or to the weight of a specific component.

## REFERENCES


1. Swanson, A. G., "A Five-Stage Solid Fuel Sounding Rocket System," NASA Memo 3-6-59L, Langley Research Center, Langley Field, Virginia, March 1959.

2. "Final Report, Project JASON," Report No. 7761-30, Aerolab Development Company, Pasadena, California, dated 5 January 1959.

3. "Systems for Space," Brochure (Unnumbered) Aerolab Development Company, Pasadena, California, dated 10 October 1960.

4. "Precision Space Systems," Brochure (unnumbered), Aerolab Development Company, Pasadena, California, undated.